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ROTOR SYSTEMS RESEARCH AIRCRAFT
PREDESIGN STUDY

FINAL REPORT
VOLUME III
PREDESIGN REPORT

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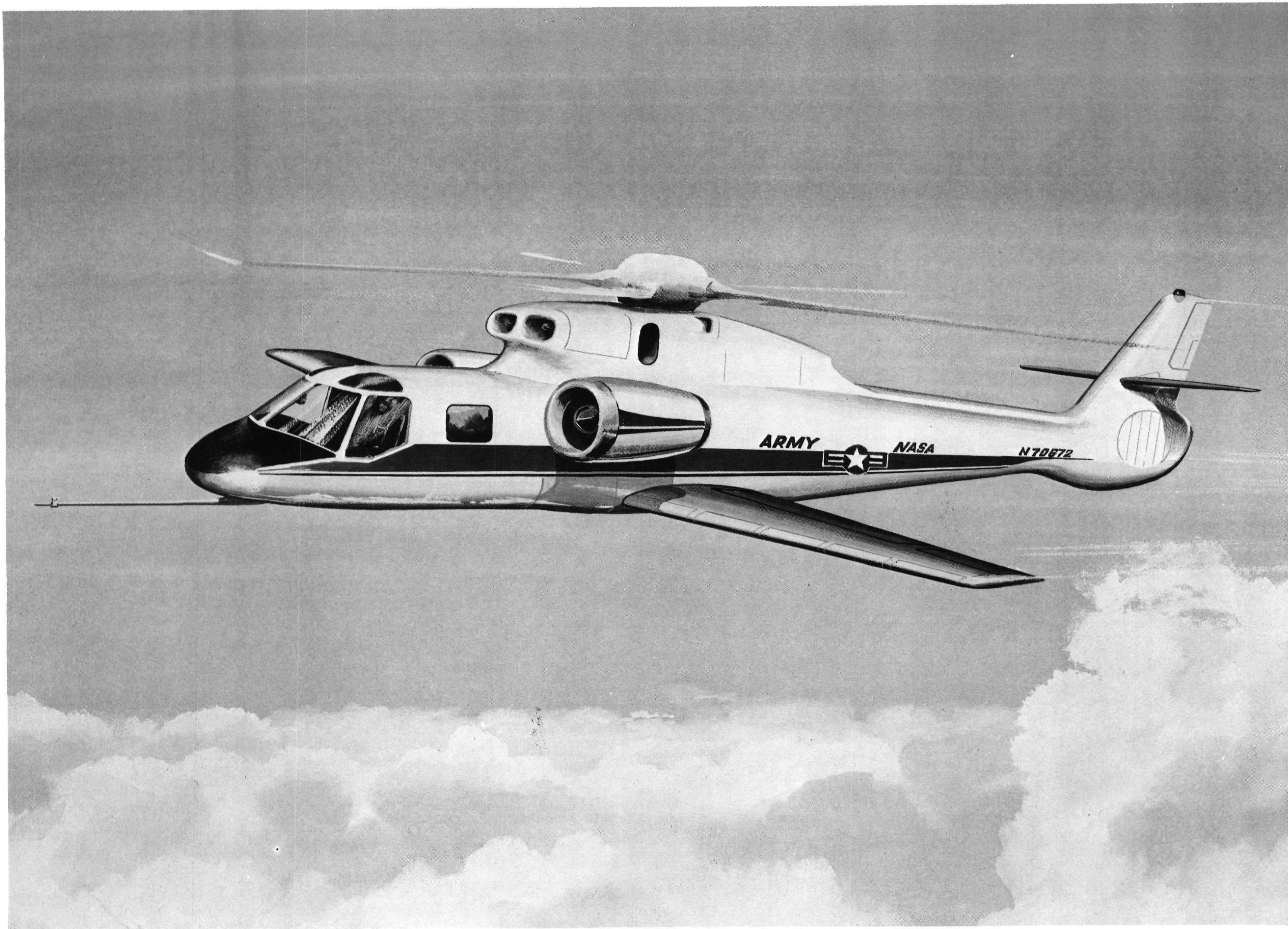
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FOREWARD

This document was prepared by Sikorsky Aircraft, a Division of United Aircraft Corporation, Stratford, Connecticut, under Contract NAS1-11228 to the National Aeronautics and Space Administration and the U.S. Army. It is subdivided into five volumes as follows:

Volume I	Summary and Conclusions
Volume II	Conceptual Study Report
Volume III	Predesign Report
Volume IV	Preliminary Draft Detail Specification
Volume V	Development Plan Report

The report covers work conducted during the period December 1971 - July 1972. Major technical contributions were made by the following Sikorsky employees:

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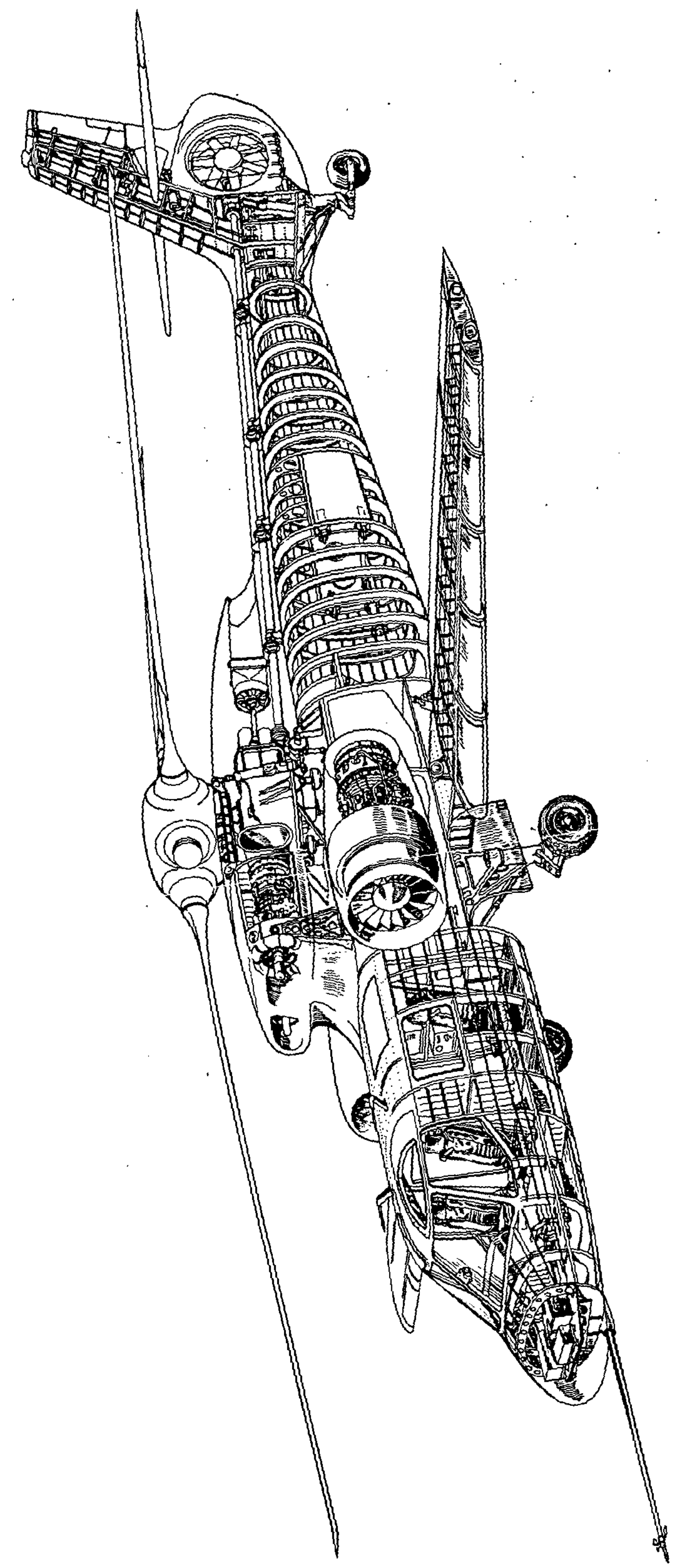


ROTOR SYSTEMS RESEARCH AIRCRAFT
PREDESIGN REPORT *

INTRODUCTION

The increasing use of helicopters by the U.S. Military services and commercial operators, and their future requirements, has created a greater need for Government research into all aspects of rotorcraft technology. Current limitations on techniques for testing and evaluating advanced rotor concepts within the Governmental agencies require that a concerted effort be made to provide the types of programs to insure a continued advancement in the state-of-the-art of rotor systems. One step which would help this advancement would be to provide an improved rotorcraft research vehicle. Recognizing this, NASA and the U.S. Army contracted for studies to define the most feasible research aircraft configuration for use by the Government at the NASA/Langley Research Center in performing the required research at minimum total program cost and in a timely manner. A secondary objective of the studies was to identify component research or technology developments, that if pursued in the scheduled development time, would improve the research capabilities of the rotor research test vehicle. The Predesign Study for a Rotor System Research Aircraft (RSRA) has been conducted by Sikorsky Aircraft between December 1971 and July 1972. The study was conducted in three parts.

Part I of the Study was concerned with determining the overall feasibility of the technical requirements and concepts envisioned by the Government for the RSRA. Engineering trade-off studies were performed to determine the desirability of any changes or additions in the aircraft requirements to minimize program time and cost without reducing capability. Two potential aircraft designs were developed to meet the requirements. One of these was an all new aircraft specifically designed as an RSRA vehicle. The second used existing aircraft components wherever feasible to reduce aircraft cost.

Part II of the Predesign Study was involved with further preliminary design of the two aircraft, including preliminary development plans and costs. At the beginning of Part II, the Government modified the aircraft technical requirements to reflect the results of the Part I study, and the designs were changed accordingly. With the conclusion of Part II, the Government selected the features of the two aircraft designs to be included in the single RSRA configuration studies in Part III. Parts I and II are documented in Volume II, the Conceptual Design Report.

Part III of the Study was involved with the further analysis of this one aircraft configuration. It included further preliminary design and a more detailed analysis of development plans and costs. Part III also included an analysis of foreseeable technical problems and risks, identification of parallel research which would reduce risks and/or add to the basic capability of the aircraft, and a draft aircraft specification. This volume documents the Part III results. With the conclusion of this study, the Government has a detailed definition of a Rotor Systems Research Aircraft, with a development plan and projected costs.

* The contract research effort which has led to the results in this report was financially supported by USAAMRDL (Langley Directorate).

RSRA AIRCRAFT DESCRIPTION

Aircraft Design Requirements and Goals

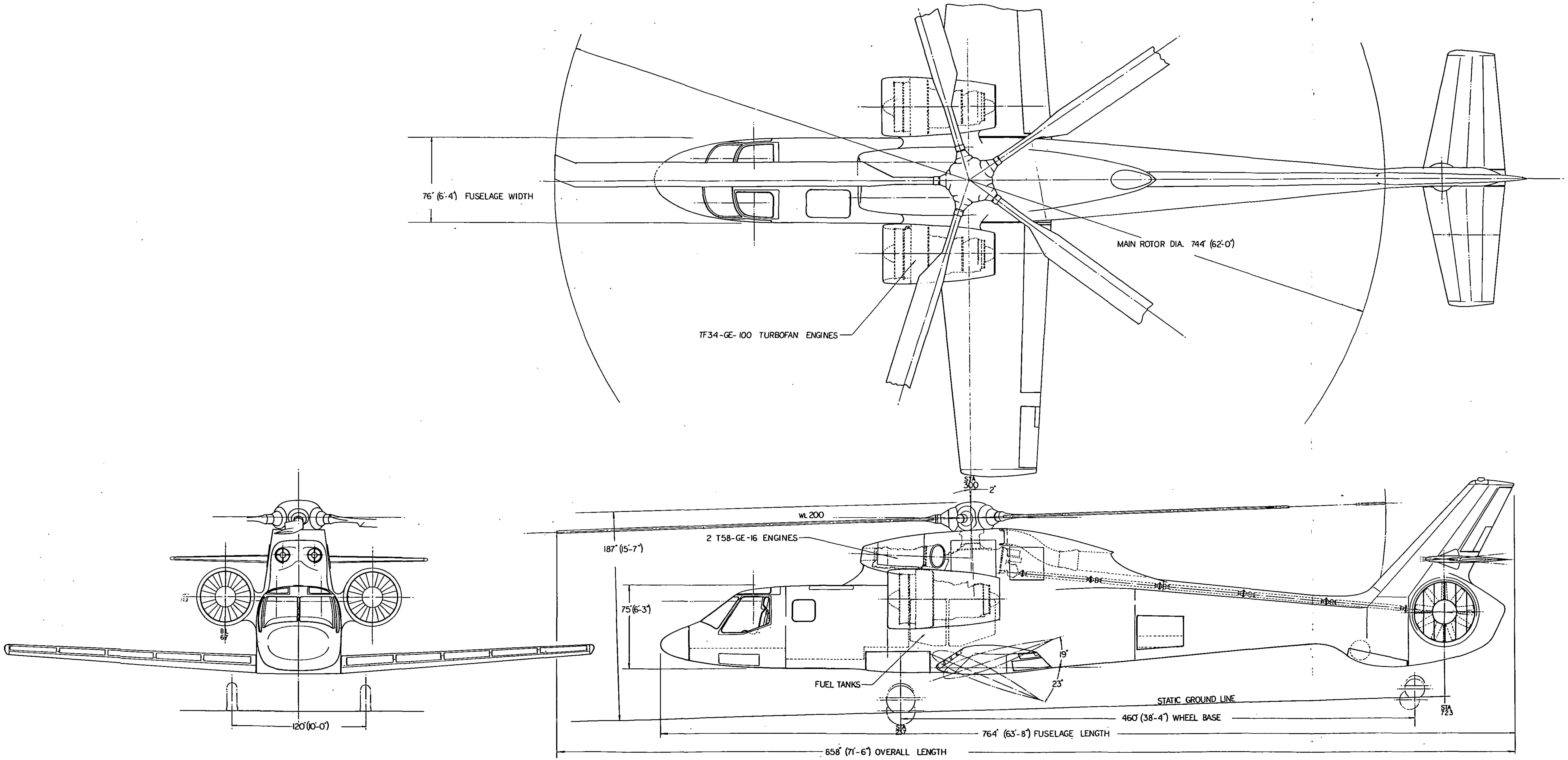
The Rotor Systems Research Aircraft was designed to achieve the requirements and goals which are thought to be highly desirable, within the state-of-the-art, and most cost effective from a rotor research point of view. From the statement of work for the RSRA study and the work performed in Parts I and II, the following list of features were selected as design requirements for the aircraft.

- Payload of 2000 pounds
- Mission fuel load for 15 minutes at 300 knots
- Fuel capacity for 30 minutes at 300 knots
- Inflight variable wing incidence
- High lift devices for completely unloading the rotor at 120 knots
- Inflight variable drag device
- Rotor force and moment measurement system
- Small ground adjustable shaft tilt
- Upward crew escape system
- Ballast system
- Design limit load factor of 4.0, ultimate load factor of 6.0
- Provisions for third crewman
- Low noise levels
- Fixed wing landing gear and braking requirements
- Independent fixed wing and rotor control system
- Onboard computer capable of model following inputs to the control system
- Wing force measurement system
- Antitorque system thrust and power measurements
- Auxiliary thrust measurement
- Capability of accepting new and different rotors for future application
- Capability of being mounted in the Ames Wind Tunnel

Applying the criteria and features quoted above resulted in the following aircraft design.

The RSRA Description

The Rotor Systems Research Aircraft is a combination of all new and existing components chosen for maximum aircraft flexibility and minimum cost. Major all new components include the basic aircraft fuselage, empennage and wings. The existing components include an S-67 main rotor, a roller main gearbox, two TF34-GE-100 turbofan engines, two T58-GE-16 engines, and a tail fan from the U.S. Army Fan-in-Fin Program. A general arrangement of the aircraft is shown as Figure 1. Use of the existing components minimizes the program cost.



MAIN ROTOR

DIAM.	62'	
BLADES	5	
CHORD	1.52	
AIRFOIL	NACA 0012	
TIP SWEEP	20°	
TWIST	-3°	

FAN DATA

DIAM.	4'-8"	
NO BLADES	7	
A.F.	950	

WING DATA

	LARGE	SMALL
AREA	348	184
SPAN	45'-8"	33'-3"
AR	6	6
FLAP AREA	42.8	40
C_T	5'-6"	3'-10"
C_R	9'-3"	6'-5"
AIRFOIL	NACA 23015	
DIHEDRAL	3°	3°

FIGURE 1

Making all new components in the critical areas provides the necessary research capability. A list of attributes of the aircraft is shown in Table I. An inboard profile showing the major subsystem components is presented in Figure 2. Descriptions of these subsystems follow.

TABLE I
AIRCRAFT DESIGN PARAMETERS

Design Gross Weight	26392 lbs
Empty Weight	
Hover Mission: No wing, No Aux. Propulsion	15599 lbs
"Helicopter Simulation" Mission: Large Wing, Aux. Propulsion Installed	21925 lbs
High Speed Mission: Small Wing, Aux. Propulsion Installed	20559 lbs
Fuel Weight, 300 knot mission	3313 lbs
Vertical Drag, Large Wing Installed	6.72%
Disc Loading, at design gross weight	8.74 psf
f, Small Wing Installed	23.6 ft^2
Ultimate Vertical Load Factor	6.0 g
Main Rotor	
Radius	31 ft.
Chord	1.52 ft
Tip Speed (Hover)	686 fps
C_T/σ (Hover @ SLS)	.0795
Twist	-3°
Number of Blades	5
Tail Fan	
Radius	2.33
Number of Blades	7
Tip Speed	726 fps
Rotor Propulsion Engines	
Number	2
Type	GE-T58-16
Military Power	1870 HP
Auxiliary Propulsion Engines	
Number	2
Type	TF34-GE-100
Intermediate Installed Static Thrust	7770 lbs
Intermediate Installed Thrust at Sea Level Standard, 300 knots	5080 lbs
Drive System Design Power	3700 HP
Performance	
Design Hover	Exceeds Requirements
Dash	SLS
Dash Speed, Small Wing	309 Knots
Dash Speed, Large Wing	305 Knots
Horizontal Tail Area	90 ft
Vertical Tail Area	50 ft^2
Wing Area, Large Wing	348 ft^2
Wing Area, Small Wing	184 ft^2

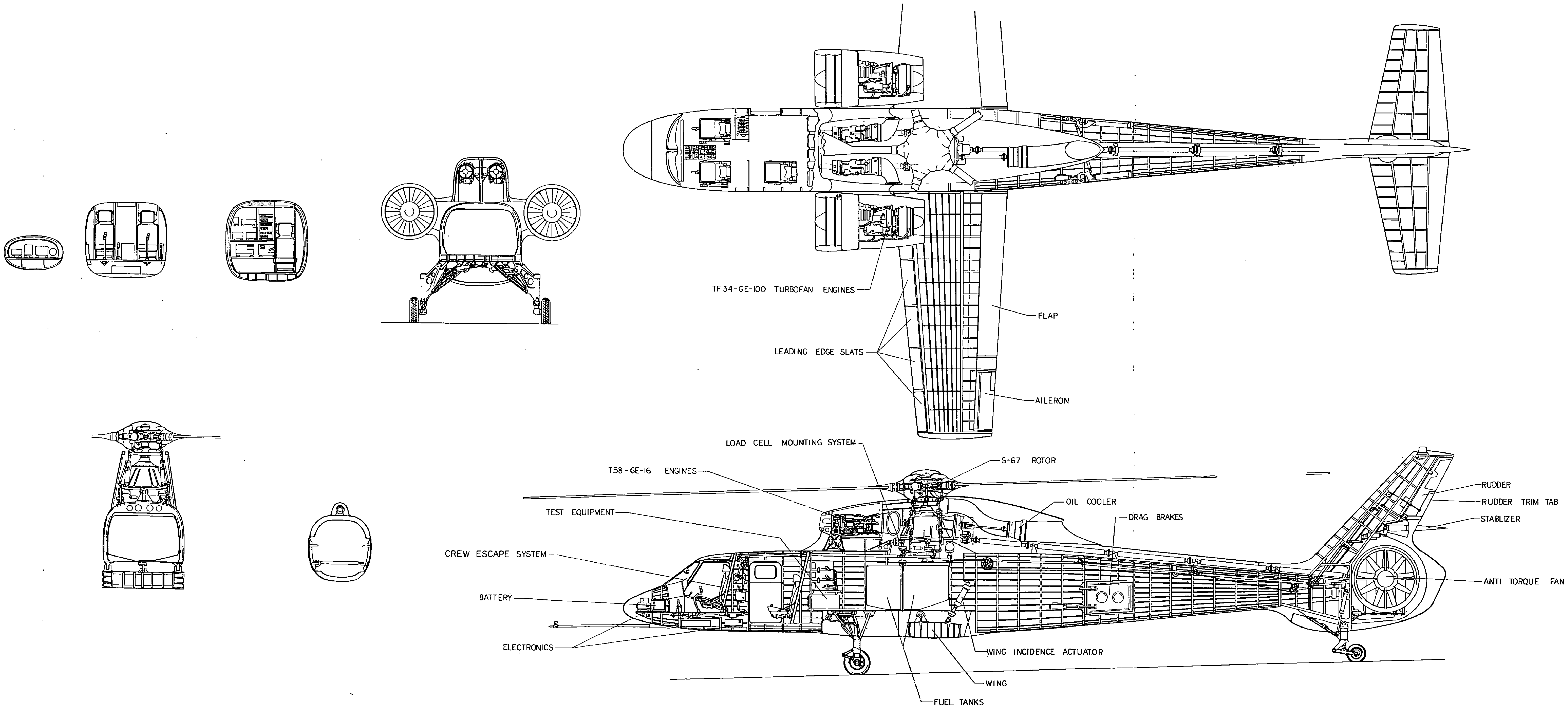

TF 34-GE-100 TURBOFAN ENGINES
FLAP
LEADING EDGE SLATS
AILERON
LOAD CELL MOUNTING SYSTEM
S-67 ROTOR
T58-GE-16 ENGINES
OIL COOLER
DRAG BRAKES
TEST EQUIPMENT
CREW ESCAPE SYSTEM
BATTERY
ELECTRONICS
RUDDER
RUDDER TRIM TAB
STABLIZER
ANTI TORQUE FAN
WING INCIDENCE ACTUATOR
WING
FUEL TANKS

FIGURE 2

Main Rotor System

The RSRA main rotor is a 62 foot diameter S-67 rotor. It has five blades and a 1.52 foot chord. The twist is -3° and an 0012 blade section is used. The last seven percent of the blade is swept aft 20 degrees relative to the span axis. Aft tip sweep and low twist were selected to obtain low vibratory control loads, low blade stresses at the high RSRA forward flight speeds, and improved hovering efficiency through compressibility relief. A drawing of the rotor blade is shown as Figure 3. The blades incorporate BIM - the visual blade inspection method used on all Sikorsky blades.

The rotor head of the RSRA is essentially the same as the U.S. Navy SH-3D helicopter with the deletion of the blade folding components. A rotor head fairing is employed similar to the S-67 Blackhawk helicopter to reduce rotor head parasite drag. Flap and lag hinges are located 12 5/8 inches from the center of rotation. The rotor head is shown as Figure 4.

The steel upper plate and splined hub are forged integrally, and the lower forged plate is bolted to the flange at the lower skirt of the splined hub. The blade pitch change bearings, attached between the sleeve and spindle, flap and lag with the blade. The vertical hinge, which permits blade lag motions, rotates on needle bearings mounted in the lower plate and tapered roller bearings in the upper plate. The vertical hinge also houses a set of needle bearings which allow flapping around the horizontal hinge pin of the sleeve-spindle assembly. Linear hydraulic dampers attached to the extension of the horizontal hinge pin provide lag damping.

The main rotor sleeve rotates about the spindle on a ball bearing stack assembly to provide blade pitch freedom. The sleeve is controllable through a horn assembly secured to the sleeve. The blade is attached through the cuff to the outboard face of the sleeve by ten high strength tension bolts. A centrifugally operated droop restrainer, attached to the spindle and vertical hinge extension, prevents excessive blade droop with the rotor shutdown.

The RSRA main rotor blades can be removed easily from the main rotor head sleeves by removing the ten high strength tension bolts. Removal of the one-half inch diameter bolts requires only a standard socket wrench and installation requires a 100 foot-pound capacity wrench. This will facilitate quick main rotor blade changes to new planform/tips, twist, etc. with a minimum of installation time. There is an amount of configuration variation possible without requiring a different set of main rotor blades. The blades are made with an extruded aluminum spar and small changes in radius and/or tip shapes can be made by cutting off the outboard portion of the spar. The pockets are bonded onto the spar and minor changes in blade chord can be made by changing the pockets.

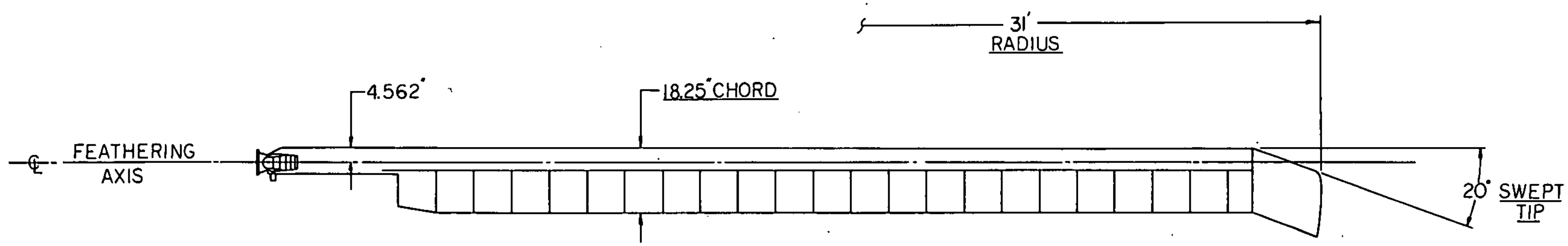


FIG. NO. 3
R.S.R.A. MAIN ROTOR BLADE

TWIST -3°

Rotor Drive System

The rotor drive system transmits power from two General Electric T58-GE-16 engines through the main gearbox to the main rotor shaft, yaw fan and accessories. Figure 5 is a drawing of the major sections of the drive system.

The main gearbox is rated for 3700 HP and has an overall reduction ratio of 93.2 to 1, reducing the engine output speed of 18766 rpm to 203 rpm at the main rotor. The two input drive shafts connecting the engines to the gearbox incorporate an electric impulse torque monitoring system to measure the output torque of each engine. The shafts are connected to the input spiral bevel pinions through a crowned spline coupling. The 3.05 to 1 spiral bevel first reduction stage rotates the drive through 86° so that the output bevel gear shaft is parallel to the main rotor. Within the output bevel gear shaft is an overrunning cam roller type freewheel unit. The cam output of the freewheel unit drives the pinion gear of the second stage spur reduction gearset. The second stage of gearing has a reduction of 1.54 to 1. It combines the power from the two inputs onto a shaft where the centerline is common with the main rotor shaft. Spiral bevel gears transmit power from this shaft to the tail take-off drive and accessory section of the gearbox. The remaining power from the combining shaft is transmitted to the final stage, the Roller Gear Drive unit, which has a reduction ratio of 19.85 to 1. The unit consists of a sun gear input, two rows of compound planetary gears with seven pinions per row and a ring gear output. The split power path originating at the ring gear induces symetrical loading at each mesh in the Roller Gear Unit. This design eliminates planet bearings except in the last row where spherical roller bearings are used to react the torque, and ensure parallel alignment of all elements within manufacturing tolerance. The roller gear drive unit has inherently more stable load sharing characteristics than conventional planetarys due to the accurate positioning of the pinions by the rollers. The housings and covers for the gearbox are cast from ZE-41C, a high strength magnesium alloy. Pads for mounting the rotor head servos are cast integral with the main housing. Bolted onto the rear of the main housing is the rear cover assembly which contains the drives and mounting pads for the various accessories.

The drive for the yaw fan incorporates a multiple disc flexible coupling between the tail take-off flange and drive shaft. This coupling will accommodate the angularity between the shafts that results from the various ground adjustable tilt angles the gearbox will assume. The main housing is bolted to a spacer plate to properly locate the load cells. The tail drive shaft is supported by viscous damped bearings at lengths determined by critical speed calculations. The tail drive shaft is connected to the right angle gearbox integral with the fan-in-fin.

Cooling System

The main transmission cooling system is an air-oil system consisting of a heat exchanger and blower assembly. This assembly is located behind the main transmission with the blower fan driven by a shaft from the accessory section.

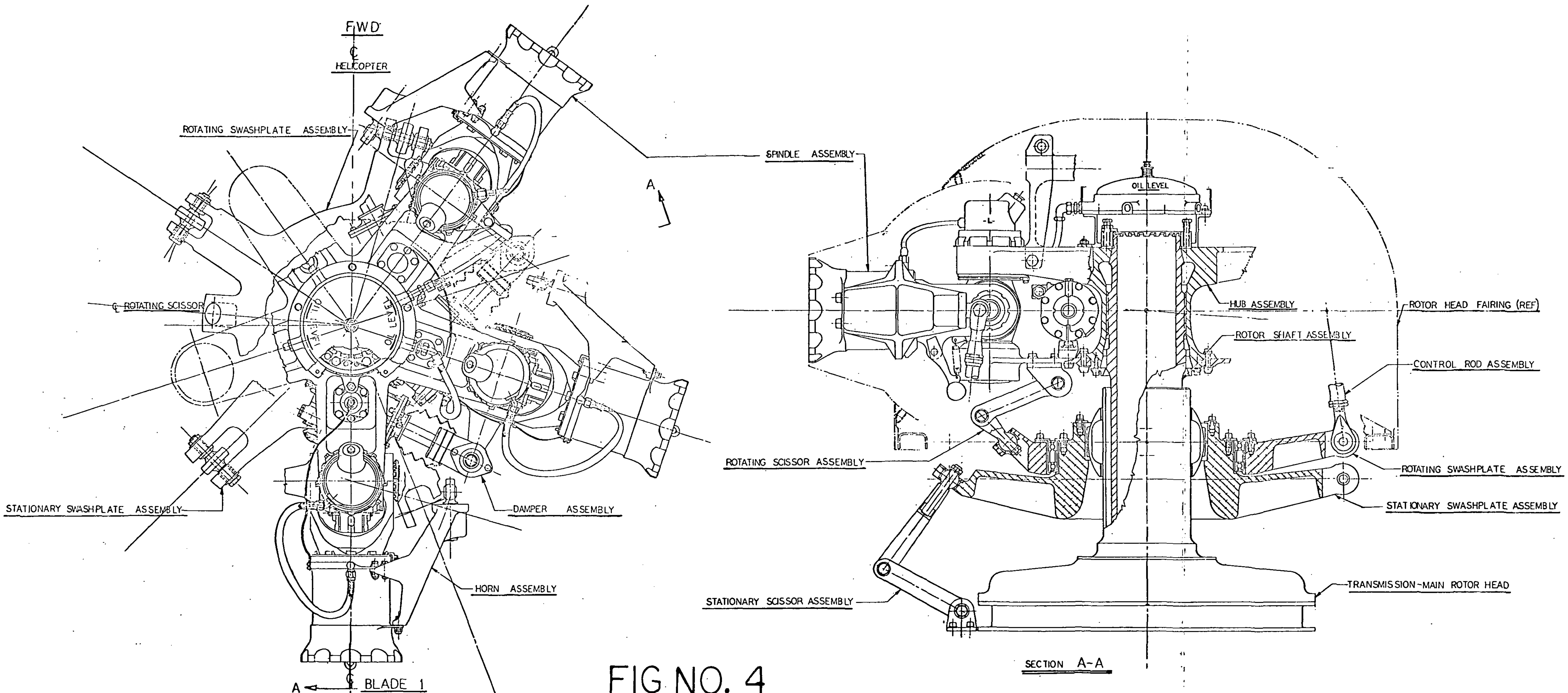


FIG NO. 4
RSRA – ROTOR HEAD – (S-67)

Lubricating System

The oil lubrication system of the main transmission contains redundant dual lubrication pumps. The number one pump is located in the gearbox sump and the number two pump is located on the rear cover accessory pad. The vane pumps used are extremely tolerant to contamination and each is capable of supplying complete lubrication required for the transmission. These pumps deliver filtered and cooled oil to the bearings and gear meshes. The oil filter has a built in by-pass warning indicator that can be reset only after the removal of the filter.

Malfunction Detection System

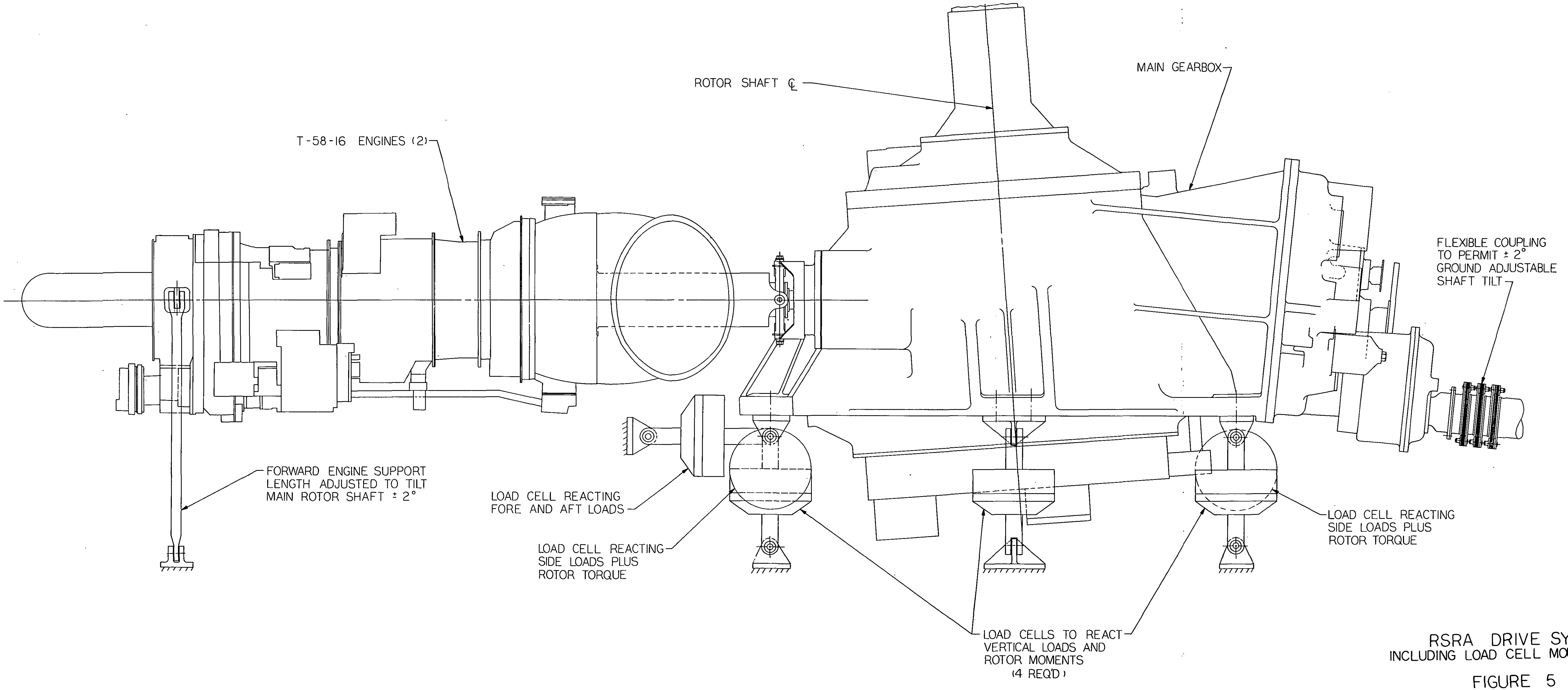
The main transmission incorporates four malfunction detection probes; one in each input section, one in the accessory section, and one in the gearbox sump. Lights on the pilots' consoles illuminate in the event of an impending malfunction or excessive wear.

Additional main transmission sensors include oil system pressure, low oil pressure warning, oil in and oil sump temperatures.

Overrunning clutches used in the main gearbox are a well proven Sikorsky ramp roller design. One per engine input is used. These overrunning clutches prevent the transmission input shaft and the engine free turbine section from being driven during single engine operation and auto-rotation.

Anti-torque System

The anti-torque system for the RSRA aircraft incorporates the fan-in-fin which Sikorsky is presently developing under a U.S. Army-funded program . The system is mounted in the aircraft empennage below the horizontal tail such that there is no angle change intermediate gearbox required. Figure 6 shows the installation on the S-67 aircraft which is being used as a test aircraft in the present program. An existing tail rotor servo is used for fan pitch control, and a fan pitch-to-rudder gain and bias control mechanism is included. The fan is mounted on load cells for thrust measurement.



RSRA DRIVE SYSTEM
INCLUDING LOAD CELL MOUNTING SYSTEM

FIGURE 5

An analysis was conducted to check the capability of this fan on the RSRA aircraft. The analysis consisted of comparing the thrust and power requirements of the fan in the RSRA and the S-67. Since the RSRA critical design point is sea level, 95° and the S-67 critical point is 4000' 95°, the higher gross weight of the RSRA is somewhat neutralized by the lower design density altitude. The study compared the two configurations against the one inch input requirements of MIL-8501A; the RSRA C_T/σ operating point is only 5% more than the S-67. This is well within the capability of the fan. The power requirement increase is about 35%, but is still well within the capability of the fan and the fan gearbox. With the T58-GE-16 engines installed, enough power is available to the RSRA fan.

Cockpit General Arrangement

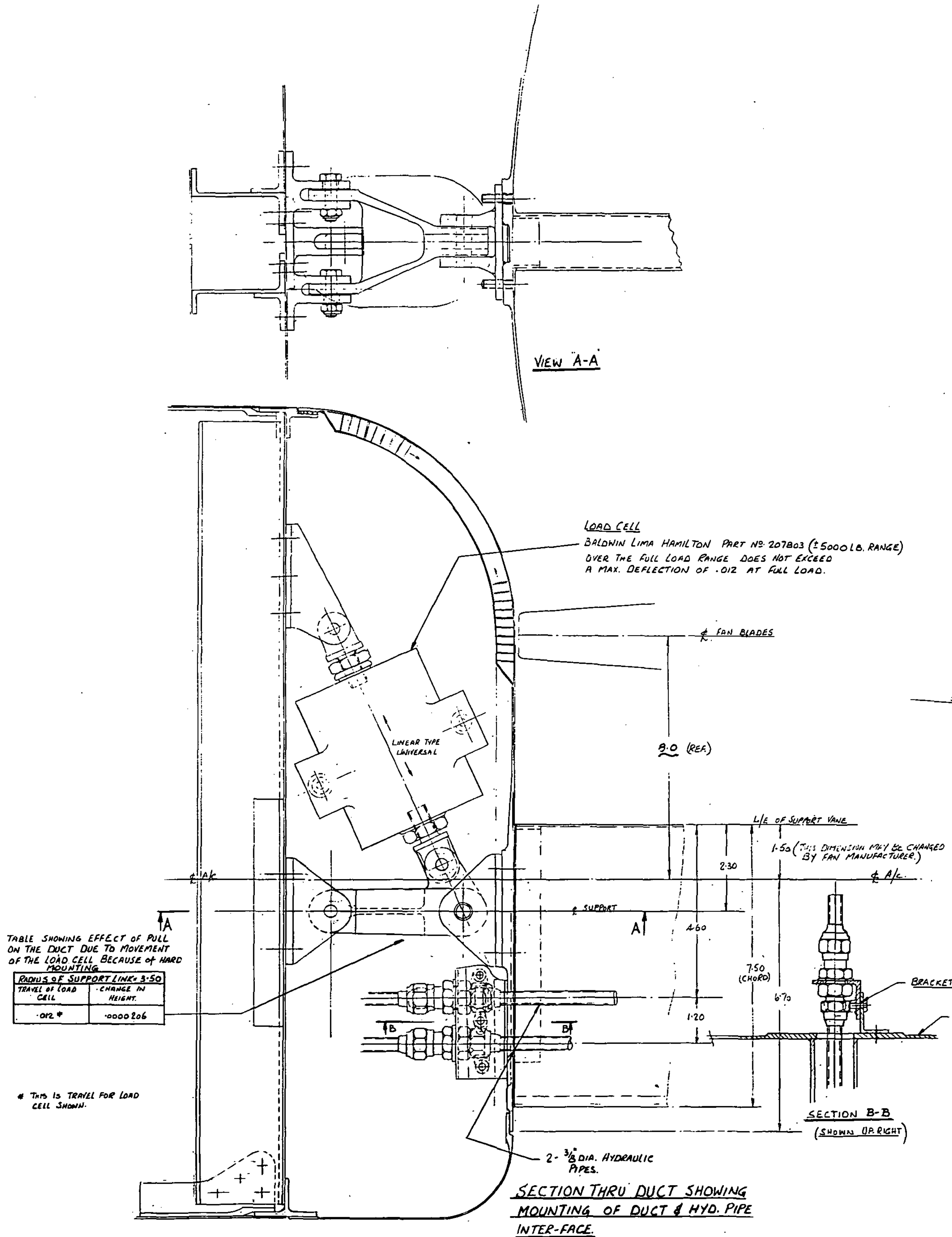
The RSRA cockpit provides side-by-side seating with pilot positioned on the right and copilot on the left. Each pilot is provided a Stanley Aviation "Yankee" escape system in the event an inflight emergency is incurred requiring immediate extraction of the flight crew. Cockpit design adheres to the guidelines of MIL-STD-1333 and MS33575. Controls location and actuation is derived from the requirements of MIL-STD-250C. Human engineering design criteria and human compatibility requirements are drawn from pertinent sections of MIL-STD-1472. External visibility from pilot/copilot seated positions generally conforms to the requirements of MIL-STD-850. A plot of pilot visibility is provided in Figure 7 using the equal area projection techniques described in MIL-STD-850.

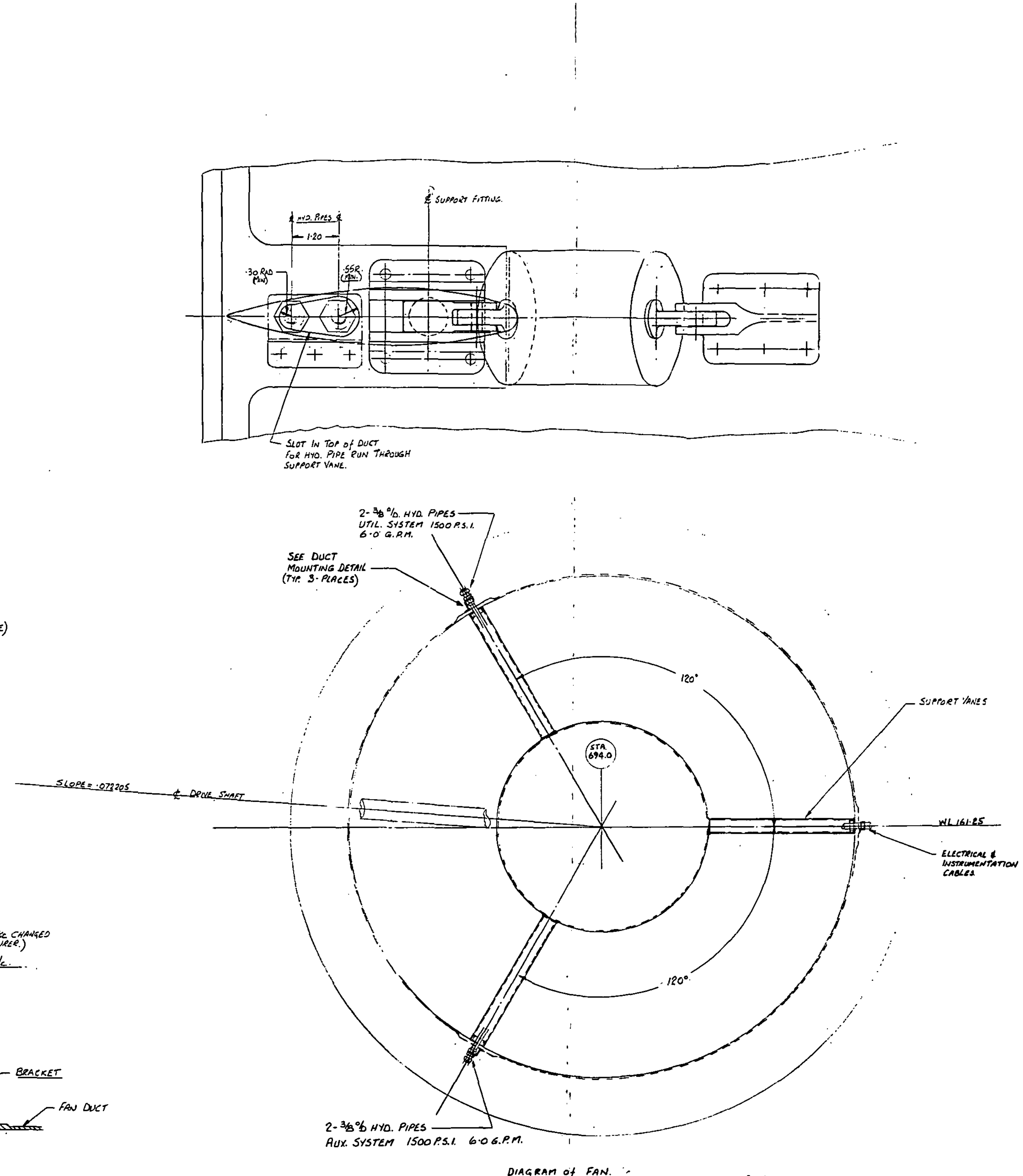
Normal cockpit entry and exit is made through the cockpit access between the seats leading to the main entry door located on the right side of the ship immediately aft of the cockpit bulkhead. Emergency egress on the ground is provided by jettisoning the large side window hatch outboard of each pilot or by jettisoning the upper escape hatch above each pilot.

Cockpit Primary Flight Controls

Primary flight controls for pilot and copilot consist of cyclic control stick, collective control stick and yaw pedals. Pilot cyclic and collective control stick, though conventional in design and function are linked with the aircraft control systems through an electrical control interface. Copilot cyclic and collective controls are conventional in design, and function through a conventional mechanical interface. Pilot and copilot cyclic and collective are not mechanically linked. Both pilot and copilot cyclic stick grips feature thumbwheel trim controls for trimming the Force Augmentation System (FAS).

Both pilot and copilot yaw controls are conventional mechanical designs which are mechanically linked. Pilot and copilot seats and controls are centered on butt line 20. Control position limits, operational envelopes and neutral reference positions are defined by MS-33575 and MIL-STD-1333. Pilot and copilot collective sticks feature dual twist grip controls for maintaining continuous thrust control for each TF-34 turbofan engine. The arrangement permits both selective and ganged control of these engines.





ANTITORQUE FAN
FIGURE 6

Control Panel Arrangements

The RSRA cockpit features a two-bay center console arrangement between the pilots and a single bay overhead panel on the aircraft centerline. The cockpit display diagram is shown as Figure 8. The center console, adjoining the instrument panel at the Cathode Ray Tube (CRT) installation contains the following control panels in descending order:

- Flight Configuration Control Panel, AFCS control, fuel management control, pilot and copilot ICS control panels, ARC-116 UHF-AM control, ARC-115 VHF-AM control, APX-72 IFF transponder control, and the ASN-43 compass control.

  The overhead panel contains the power quadrant for control of the T-58 turboshaft engines and, in order of their position aft of the quadrant, the following:

  - The fire emergency control panel, master switch control panel, exterior and interior lighting control panel, and essential circuit protective devices

  Engine fire bottle arming controls for all four engines incorporating illuminated fire warning capsules in Tee-handles are located under the center portion of the instrument panel glare shield readily accessible to both pilots. The landing gear control, incorporating emergency gear extension, is located in the lower instrument panel below the CRT display. The landing gear position indicator is adjacent to the control handles. Master caution capsule annunciators are incorporated above each pilot's VGI.

Miscellaneous Equipment

Miscellaneous equipment provided in the RSRA cockpit shall consist of one fire extinguisher and one first aid kit located on the cockpit bulkhead.

Cockpit Instrumentation

The RSRA cockpit provides primary flight instrumentation for both the pilot and the copilot and shared displays to monitor propulsion and critical aircraft subsystems. The centrally located CRT display is provided enabling either pilot monitoring and control capability of flight limiting test parameters.

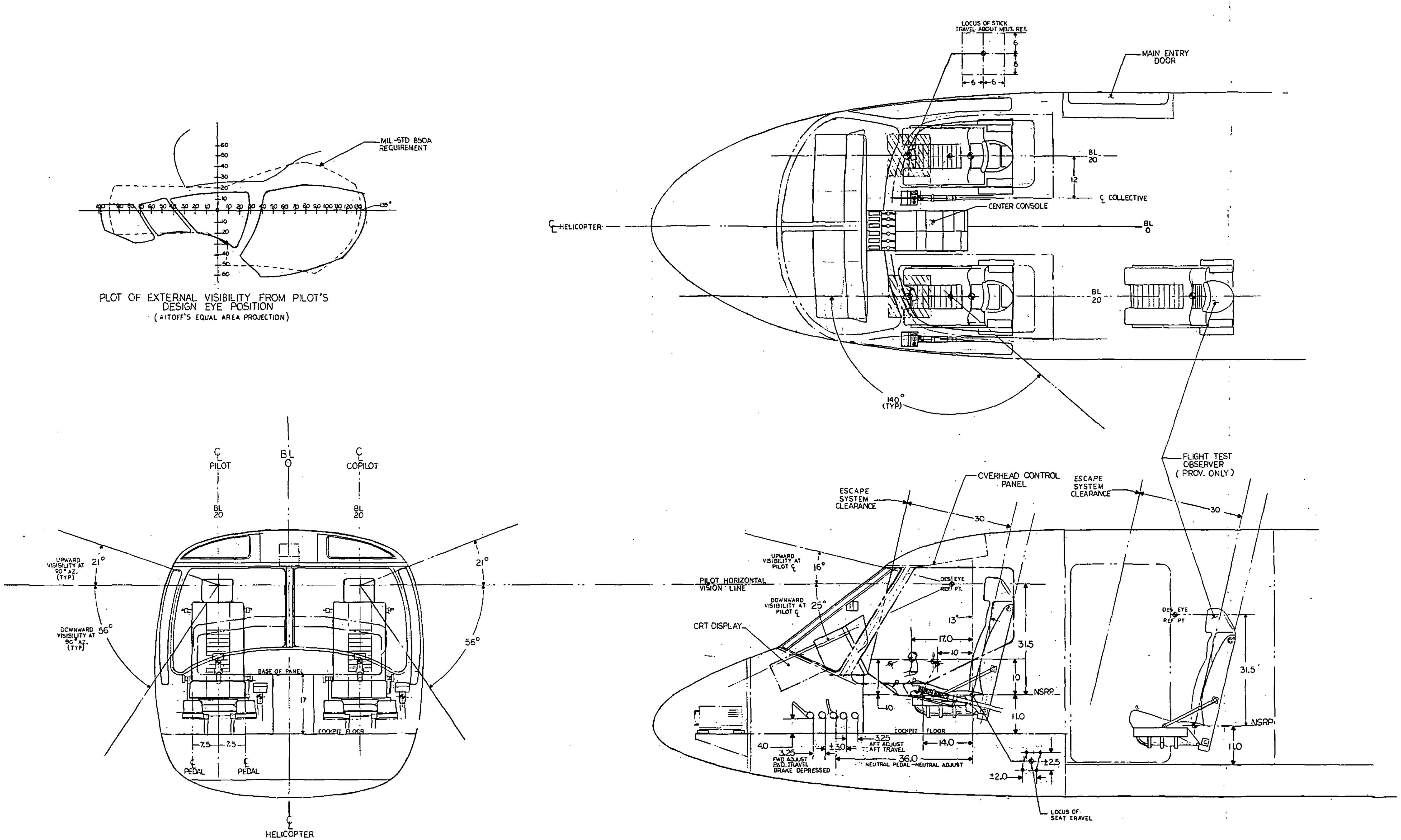


FIGURE 7

RSRA
COCKPIT GENERAL ARRANGEMENT
& VISIBILITY DIAGRAM

Primary Flight Instrumentation

Pilot and copilot primary flight instrumentation is grouped per the requirements of MS33785. Each display consists of vertical gyro indicator (incorporating turn and slip reference), bearing-distance-heading indicator, airspeed, barometric altimeter (pilot's indicator incorporating altitude reporting encoder), instantaneous vertical speed indicator, dual torquemeter, triple tach indicator, (displaying N_{F_1}, N_{F_2}, and N_R), and elapsed time clock. Additionally, the pilot's display includes a course deviation indicator (providing ILS/NAV information), a G-meter, and blade tip mach indicator. A standby compass and outside air temperature gauge are centrally located above the instrument panel.

Engine and Power Train Instrumentation

Engine and power train displays are centrally located for monitoring by either pilot. The displays are vertical scale type combining compactness and light weight in a system that can be monitored quickly and accurately. Those instruments providing reference to the T-58 turboshaft engines include gas generator tach indicator (Nf with integral digital readout), power turbine inlet temperature (T_5 with integral digital readout), fuel flow, engine oil pressure and engine oil temperature. Power train and aircraft system displays include main transmission and tail fan gearbox oil pressure and oil temperature, hydraulic pressure, fuel quantity, and hydraulic oil quantity.

Similar displays provided for TF-34 Turbofan engine reference include gas generator tach (Ng with integral digital readout), fan turbine tach (N_F with integral digital readout), turbine inlet temperature (T_5 with integral digital readout), engine oil pressure and engine oil temperature.

Supplementing the engine and power train system primary displays are appropriately marked illuminated capsules within the caution-advisory panel located on the instrument panel. This will alert the flight crew to critical operational levels.

Flight Test Instrumentation and Displays

Instrumentation is provided enabling either pilot or an observer to monitor all critical test parameters in progress. Test instrumentation includes side slip angle indicator, collective position indicator, cyclic stick longitudinal position indicator, cyclic stick lateral position indicator, rudder position indicator, stabilator-rotor cyclic mixing indicator, aileron-rotor cyclic mixing indicator, tail fan-rudder mixing indicator, wing flap position indicator. In addition to these standard displays, a CRT display measuring 8 inches by 8 inches by 24 inches deep is located in the top of the instrument panel centrally positioned for convenient monitoring by either flight crew member. The area immediately forward of the center console houses the flight configuration control panel containing position-related

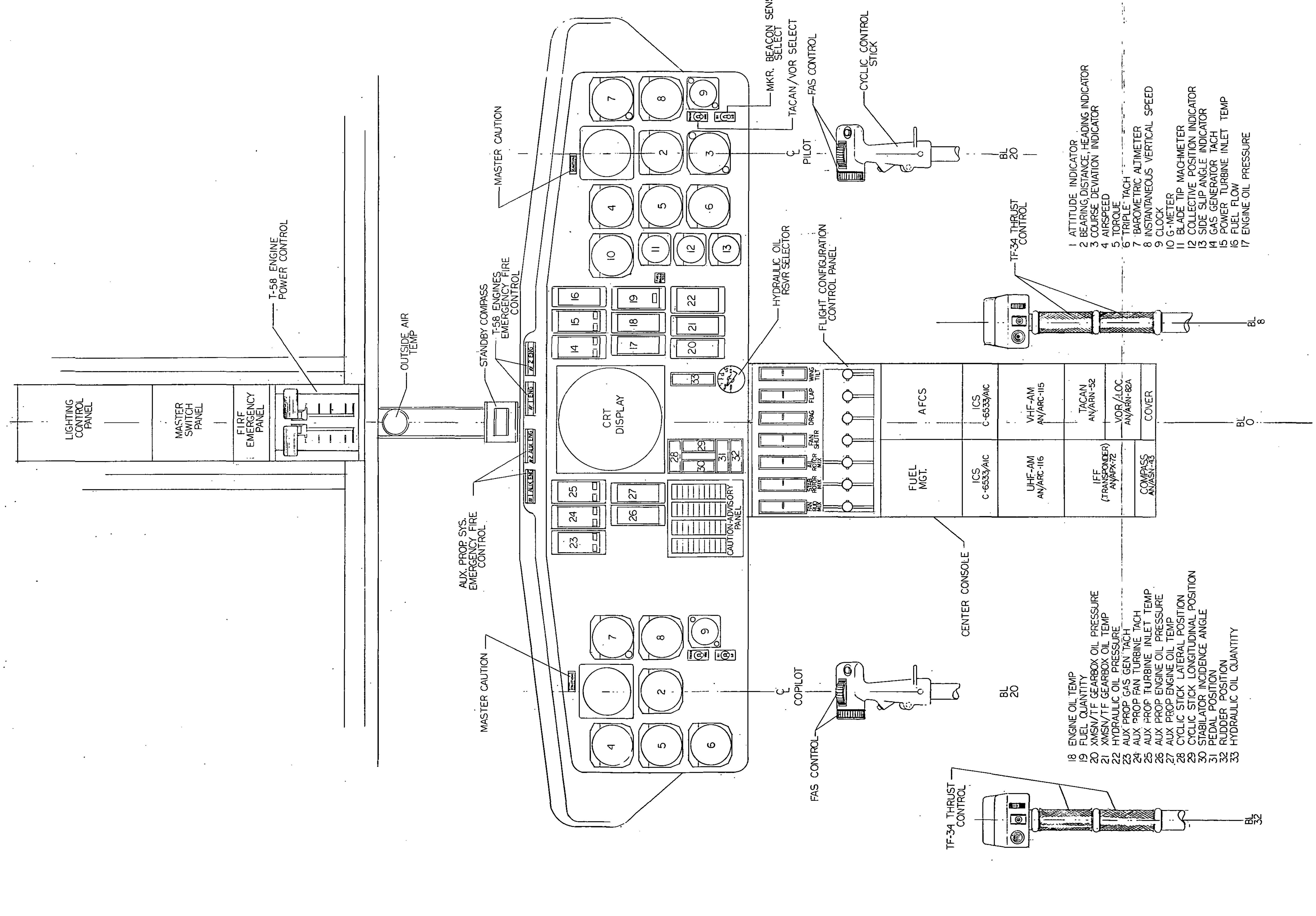


FIGURE 8

RSRA

COCKPIT DISPLAY DIAGRAM

switch controls that permit selective positioning of each control system configuration while providing immediate visual reference to the position of all other control settings. Controls and associated displays contained on this console include wing tilt position, wing flap position, drag brake position, tail fan shutter position, tail fan/rudder mixing position, stabilator-rotor cyclic mixing position, and aileron-rotor cyclic mixing position.

Crew Escape System

Emergency escape systems including extraction seats and separable capsules are already highly developed and effective systems incorporated in most combat type fixed wing aircraft. The percentage of successful escapes from disabled aircraft is very high (in the order of 90 percent) and a strong effort continues to further improve this record. These developments in crew escape canopy separation, and capsule separation could be applied to helicopters except for the interference posed by the main rotor blades.

During this RSRA Predesign Study, crew escape systems were studied which would provide the capability to abandon an uncontrollable aircraft in flight. The following requirements were established for the system:

- Zero altitude, zero speed capability
- The system must be able to operate successfully during an emergency in nap-of-the-earth testing
- The aircraft is flown by experienced test pilots at all times
- The aircraft will not be abandoned in flight unless it is uncontrollable
- Escape at maximum flight speeds is required

Escape systems considered for the RSRA were upward, downward and sideward crew escape, capsule ejection, and manual bail out. The following is a comparison of each of the systems compared to the aforementioned requirements.

Upward Escape -- meets all of the requirements but requires a blade severance system.

Downward Ejection -- This method has been used in the past for fixed wing applications and avoids the requirement for a blade severance system. It is currently in disfavor with military because it requires about 300 feet or more of altitude for safe chute opening.

Redesign of the cockpit platform structure and rerouting of flight controls to accommodate downward ejection results in an increase in aircraft weight, and added structural and flight control complexity.

Sideward Ejection -- Sideward ejection would also avoid the need of a blade severance system but would not be a zero system. Ejection forces would require turning the seat toward the side and tilting it parallel to the trajectory before rocket ignition to protect the pilot's spine from injury.

Since sideward ejection has never been operational, a special development program would be required to develop the seat.

Capsule Ejection -- meets all requirements, requires a blade severance system. It is heavier than the seat extraction.

Manual Bail Out - Does not meet requirements of crew escape with the aircraft in an uncontrollable state, as attitude and " g" forces can be incompatible with this method.

Upward escape was selected for the RSRA as it meets the requirements with a minimum weight penalty. It does require a blade severance system. However, recent tests by Sikorsky of such a system have proven its basic capability to the point where it can be considered without an undue increase in RSRA program risk.

On 14 December 1971, a sequential blade severing system developed under a Sikorsky Aircraft funded R&D program was demonstrated to the Military. This test demonstrated the feasibility of blade severance and showed that the location at which blade separation occurs can be controlled and repeated precisely. The design of the rotating transfer and sequencing mechanism is such that blade separation can be made to occur at any blade position, in any order, and/or in any combination.

The rotor blade severing system is designed such that each blade is severed just out board of its cuff by a flexible linear shaped charge which is attached externally without blade modification. The charge is detonated by pulling a handle in the cockpit which starts a confined detonation stimulus. This signal is transferred to the rotor through intermediate lines and a rotating transfer unit. As presently planned for the RSRA, the blades are severed simultaneously by a primary system with a redundant backup system designed to fire after a delay of one rotor revolution (0.3 seconds).

The blade shedding system is a fully independent system, having no connection with the aircraft's electrical or hydraulic systems. It propagates initiation from the cockpit to the rotor blades through SMDC (Shielded Mild Detonating Cord) with a chemical deflagration rate of approximately 20,000 feet per second. This pyrotechnic system was selected in order to achieve maximum reliability. It is impervious to RF, lightning, and stray voltage. Even gunfire tests with high explosive 20 mm rounds will not cause premature initiation. Deflagration is begun by pilot or copilot activation of percussion primers in the D-rings in the cockpit. Initiation continues through to a sequencing device at the main gearbox, is transferred to the main rotor shaft, and travels out to linear-shaped charges on the rotor blades.

The only modification required for the rotor is the addition of the linear shaped charges clamped around the blade spar, and provisions for the detonating chord. Because these are the only modifications required, this system will not be difficult to apply to any new rotor to be tested on the RSRA.

The Yankee escape system of Stanley Aviation Corporation has been selected for the means of upward escape as it has a proven record and results in a lighter total system. The Yankee escape system provides escape by using a rocket, attached to a parachute type harness, which is launched out of the vehicle. As the rocket is fired and the canopy section over the crew removed, the seats travel up rails to the edge of the aircraft. The seat pan drops to a vertical position and the rocket pulls the men out of the vehicle. After the men are clear of the vehicle, the escape system deploys a parachute. The yankee system has made 40 successful escapes to date.

The rotor blade severing plus Yankee escape system provides both pilots and the third crewman with a zero altitude - zero to 300 knot escape envelope. The basic technology for rotor blade severing has been demonstrated and the Yankee escape system is operational. An Army/Navy program is anticipated shortly which will demonstrate the marrying of the two systems into an operational rotary wing escape system. If this program does not mature, the system will be developed specifically for the RSRA. Cost for this complete development is included in the RSRA estimates.

The escape system included in the basic aircraft design simultaneously severs all blade at once. After a time delay, the canopy is separated and the crew is extracted. As an optional quote, a more elaborate system has also been estimated. This "dual stage" escape system is designed to include a sequential blade severance mechanism to separate the blades in a predetermined direction, as well as the capability to recover control of the aircraft after the blades have been removed. RSRA does have full fixed wing capability and this optional feature would allow recovering the aircraft even after the rotor system has been removed, and continuing flight in the fixed wing mode.

Airframe and Empennage

The RSRA airframe is designed fail safe with multiple load paths and low cost light weight, state-of-the-art construction. It is of semi-monocoque construction with skins which are primarily 2024-T3 clad sheet. Formed stringers and frames are 7075-T6 clad sheet. Forged fittings are 7075-T73 which effectively resists stress corrosion. A structural arrangement of the airframe is presented as Figure 9.

The airframe is composed primarily of the formed stringers, single curvature skins in the center section, and formed sheet metal frames. Forged fittings are used in concentrated load areas providing more direct load paths and simplifying splicing and reducing numbers of parts. Fairings are primarily honeycomb core with fiberglass or sheet metal skins for smoothness and shape retention. All openable panels are secured with over center latches.

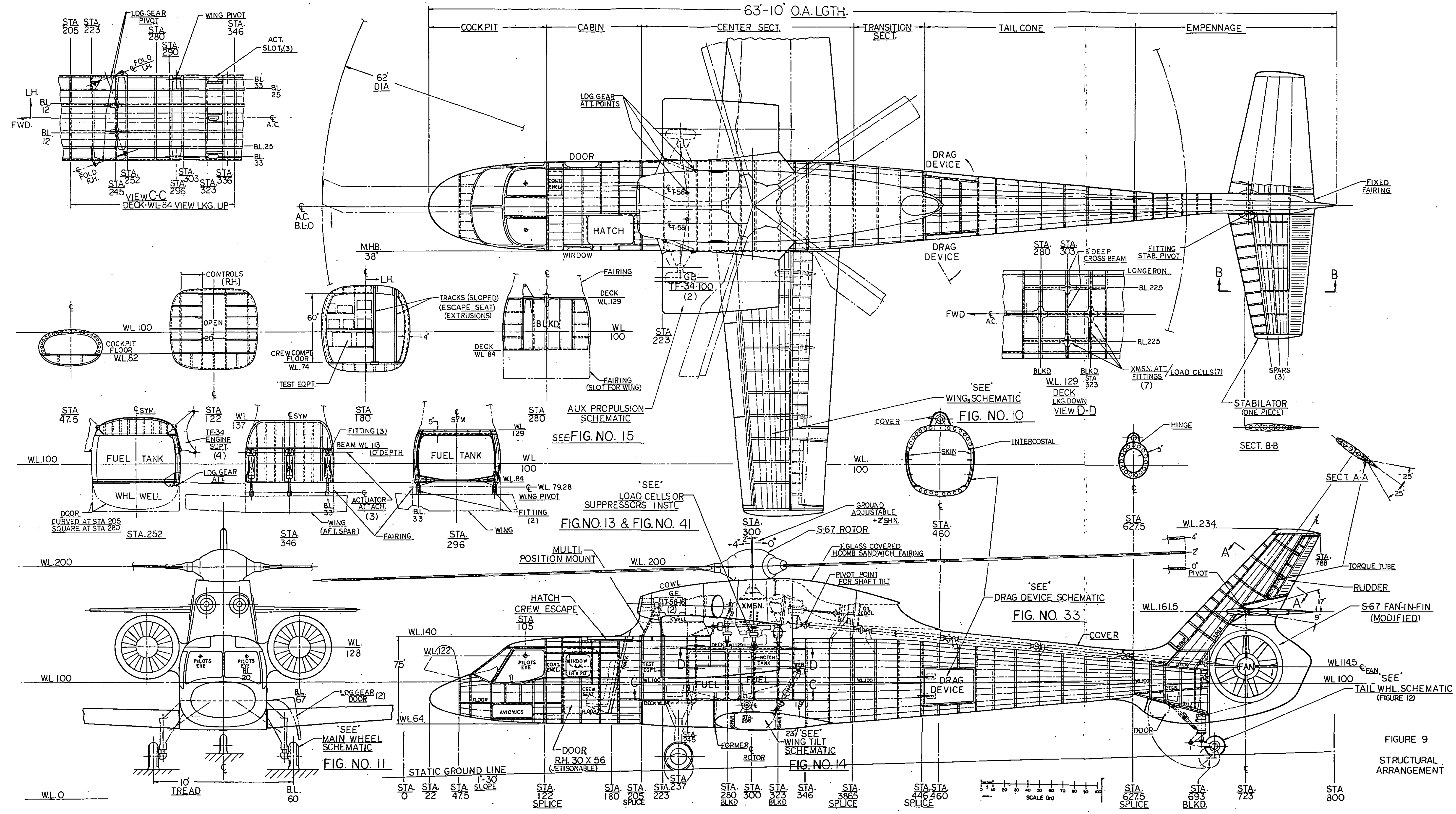


FIGURE 9

STRUCTURAL ARRANGEMENT

Cockpit - Fuselage Station (FS) 22 to 122

The cockpit is cantilevered from the cabin at FS 122. The cockpit supports pilot, copilot, escape seats, jettisonable side windows, avionics shelf, ballast compartment, flight instruments, and flight controls. Skins are flush riveted and double curved to ensure low aerodynamic drag. The cockpit tub contains seat rail beams which also act as crash skid beams.

Cabin - FS 122 to FS 205

The cabin section consists of frames four inches deep at approximately 20 inch spacing, and formed stringers at approximately 6 inch pitch, covered with clad single curvature aluminum alloy skins, flush riveted. The compartment contains an ingress-egress jettisonable door on the right hand side FS 138 to FS 168, a blow out upper hatch for crew escape, a viewing window FS 138 to 162 on the left hand side, and shelves for instrumentation. A controls enclosure FS 122 to FS 138 is provided, located behind the pilot.

Center Section FS 205 to FS 386.5

The fuselage center section contains the structure necessary to support the dynamic components, fuel tanks, landing gear, wing and auxiliary propulsion engines. This section is constructed out of aluminum alloy webs, extrusions, fittings, clad sheet and in extremely highly loaded areas, steel or titanium fittings. Bulkheads are assigned multiple tasks to minimize weight and cost and ensure an efficient structure. Skins are flush riveted and primarily single curvature. An upper deck at WL 129 is provided, supporting dynamic components, T-58 engines and accessories. The fuel tanks are cradled in their compartments with proper fume proofing, venting, line emplacement and filler location in accord with good aircraft design practices.

Transition Section FS 386.5 to FS 446

Transition section consists of four-inch deep aluminum alloy formed frames, formed aluminum alloy stringers, and double curvature clad skins. The double curvature skins allow a smooth transition from the center section to the tail cone to minimize aerodynamic drag. The skins are flush-riveted and butt-jointed with internal straps for splicing. Access ports for inspection and repair are provided.

Tail Cone FS 446 to FS 627.5

The tail cone, of conical shape, contoured and sized to minimize vertical drag, consists of floating four-inch deep formed aluminum alloy frames at 20-inch approximate pitch. Zee section through stringers are used at approximately six-inch pitch; these are covered by single curvature aluminum alloy clad sheet. Drag devices of honeycomb sandwich construction are included. These are recessed flush when closed. The necessary reinforcement is provided in this area to provide no loss of continuity of structure. Also included is structure for the ballast compartment, which is located behind the drag brakes. Hatches, access ports, etc. are provided for access and repair. The upper tail cone supports the tail drive shaft at approximately six foot intervals.

Vertical Fin FS 627.5 to FS 788

The fin consists of aluminum alloy ribs, spars, skins and stringers with forged attachment fittings. It contains a rudder, yaw fan, tail wheel, stabilator support and load cell supports for fan thrust measurement. The fin is cambered for aerodynamic efficiency and faired sufficiently for the assigned goals. Two primary spars and one rudder support spar are used. Redundancy and fail safe design practices are employed. Skins are flush riveted and butt-jointed. Fan inlet and exhaust are contoured for maximum efficiency and low drag. The fin houses the fully retractable tail wheel. The ventral fin completely encloses the tail wheel in its retracted position.

Stabilator FS 723 (Hinge Point)

The one piece, symmetrical airfoil stabilator consists of three spars, stringers, ribs and clad skins flush riveted and butt-jointed, providing both fail safe and simple construction. It is mounted at the front spar by a multi-lug aluminum alloy forging, generously sized for safety. Stops are provided to preclude breakaway.

Ballast System

The ballast system consists of two ballast bays. The aft bay is located in the tailcone in the vicinity of the drag device and the forward bay is located under the cockpit floor. Each bay has a 1000 lb ballast capacity. Ballast is in the form of 50 lb blocks of depleted uranium which are bolted, as required, to transverse support structure.

Upper Fairing

A fully flush honeycomb sandwich, with fiberglass or sheet metal covering is provided for the upper aircraft fairing. The necessary hatches, cowls, platforms, ports, inlets, etc. are provided with fail safe hinges and latches. A smooth interface between rotor head fairing and fuselage minimizes drag.

Ames Wind Tunnel Mounting

The airframe can be attached to the wind tunnel pylons by removing the main landing gear and installing a faired cross beam with appropriate knuckle fittings to bolt to the pylons. The tail wheel can be left on and a bolt-on fitting installed to accommodate the aft pylon knuckle fitting.

Drag Brakes

The drag brakes for the RSRA were sized to produce a total aircraft equivalent parasite area of 40 ft^2. The total area was chosen from historical drag charts. The charts generally show that utility helicopters with gross weights below 30,000 lbs have drags below this total. The utility helicopters had unfaired rotor heads and bulky fuselages characteristic of this type of vehicle. The equivalent parasite area which the drag devices must produce to achieve a total of 40 ft^2 is 16.4 ft^2. Selecting this value allows the RSRA aircraft to:

- Simulate any historic utility transport helicopter up to a gross weight of 30,000 lbs
- Simulate the drag of high speed helicopters at gross weights higher than 30,000 lbs
- Simulate rotor heads with over three times the rotor drag as the RSRA baseline rotor system at 26,392 lbs gross weight
- Simulate aircraft which have up to 60 percent more drag than the RSRA vehicle
- Construct "upper stall limit" charts to compare data with NASA CR-114.

The RSRA drag device location has been selected from several possible locations to achieve the following qualities:

- Minimum aerodynamic interference effects associated with brake deflection
- Maximum test flexibility
- Minimum effect on aircraft moments
- Minimum structural integration problems

Table II shows the results of a survey of six possible brake locations and sizes. The capability to achieve the desired delta area of 16.4 ft^2 was estimated based on drag coefficients for 60 degree brake deflection. Other evaluations are purely qualitative.

The split plate drag brake located on the sides of the aft fuselage is selected as it has available area, is a good drag producing device, yields good test flexibility, no undesired moments, and easy structural integration. The required brake area is 7.9 ft^2 per side.

The aircraft with small values of wetted area and low parasite drags can be simulated by using the auxiliary thrust to propel the aircraft to higher speeds.

The drag brakes have been sized to operate at the full 60 degree deflection up to speeds of 185 knots. The actuation cylinder area is 0.785 sq. in. operating from 3000 psi hydraulic system. A drawing of the drag brake actuation linkage is shown in the flight control section as Figure 33.

TABLE II
COMPARISON OF RSRA DRAG BRAKES

CONFIGURATION	CAPABILITY TO ACHIEVE 16.4 FT^2 TOTAL EQUIV. PARASITE AREA	TEST FLEXIBILITY	IMPINGEMENT ON TAIL/WING/ROTOR SURFACES	UNDESIRABLE AIRCRAFT MOMENTS	STRUCTURAL INTEGRATION PROBLEMS
1. Split plate brakes on main rotor pylon side	C_{Do} eff = 1.124 S/side 6.9 ft^2 marginal	Unable to separate drag with rotor drag at the balance system	Impinges on tail rotor	Pitch up upon deflection	Actuation in pylon
2. Split plate brakes on wing surfaces	C_{Do} eff = .63 S/side = 12.4 ft^2 available	Wing and device drag would not be independent	Possible impingement if horizontal tail is low	Pitch down upon deflection	Actuation through wing pivot
3. Split plate brakes on fuselage bottom	C_{Do} eff = 1.124 S = 13.8 ft^2 unavailable	No disadvantage	No impingement	Pitch down upon deflection	Easy integration
4. Split plate brakes on sides of aft fuselage section	C_{Do} eff = 1.034 S/side = 7.9 ft^2 (min) available	No disadvantage	Possible impingement on horizontal tail if tail is low	Essentially no moments	Easy integration
5. Split plate drag brakes on vertical tail sides	C_{Do} eff = .63 S/side = 12.4 ft^2 available	Not independent of rudder deflection	Impingement on rudder, tail rotor	Small pitch up upon deflection	Structural weight increase
6. Split plate brakes on engine pylons	16.4 ft^2 total unavailable	No disadvantage	Impingement on tail rotor	Small pitch up upon deflection	Easy integration

Wing Descriptions

Two wings are used on the RSRA. The first is a large wing for helicopter simulation from 100 to 200 knots. The second is a smaller wing for compound flight at speeds up to 300 knots.

The design of the large wing fulfills the requirement to support the gross weight of the aircraft at 150 knots, sea level, standard conditions, in a clean, unflapped configuration. The stall margin is 20 percent. An aspect ratio of six, zero sweep angle, and an 0.6 taper ratio were selected for the wing to provide the maximum lift at the design condition and also yield the best lift performance with flaps down. The unflapped wing loading is 75.8 lbs/ft^2.

The large wing is equipped with double slotted trailing edge flaps and leading edge slats. This high lift system provides the capability to unload conventional main rotors to a C_L/σ of approximately .03 at 100 knots with a 20 percent stall margin and complete unloading of the main rotor above 120 knots.

The small wing for the RSRA aircraft was designed to lower the design gross weight required for the 300 knot mission and to be more representative of high speed compound wing designs. It was designed to support 100 percent of the aircraft gross weight at 200 knots with flaps down. Plain flaps were selected to keep wing complexity and weight to a minimum. The small wing for the high speed compound testing and the large wing for helicopter simulation meet the RSRA requirements as modified for Part II for 300 knot compound testing and helicopter simulation between 100 and 200 knots. With two TF-GE-100 turbofan engines installed, speeds of the aircraft exceed the 300 knot requirement with either the large or small wing installed. Drawings of the large and small wings are shown as Figure 10.

The primary wing structure consists of two spar torque boxes with skin and Z section stringer upper and lower surfaces. The material used is 7075-T6 for the ribs, stringers, spars and other internal structure. The skins are 2024-T3 clad sheet, flush riveted and butt jointed. For concentrated load areas, aluminum alloy forgings (7075-T73) are utilized. The ailerons are push rod actuated. The flap and slat sections are interconnected for symmetric motion. Both systems are actuated by screwjacks.

The wings of the RSRA have inflight variable incidence. The incidence is varied by three hydraulic actuators which are controlled by a lever in the cockpit. The actuator range is designed to provide the full incidence range required by the wing to achieve ±10 degrees of effective rotor shaft tilt by varying fuselage incidence. The total actuator range is 42 degrees.

The wing tilt mechanism is capable of withstanding 4.0 limit load factor on the wing with the flaps retracted at speeds up to 360 knots dive speed. With flaps fully extended, the tilt mechanism can operate up to a maximum speed of 175 knots, with the wing operating up to $C_{L_{max}}$. The cylinder area is 14.1 sq. in. per cylinder operating from a 3000 psi hydraulic system.

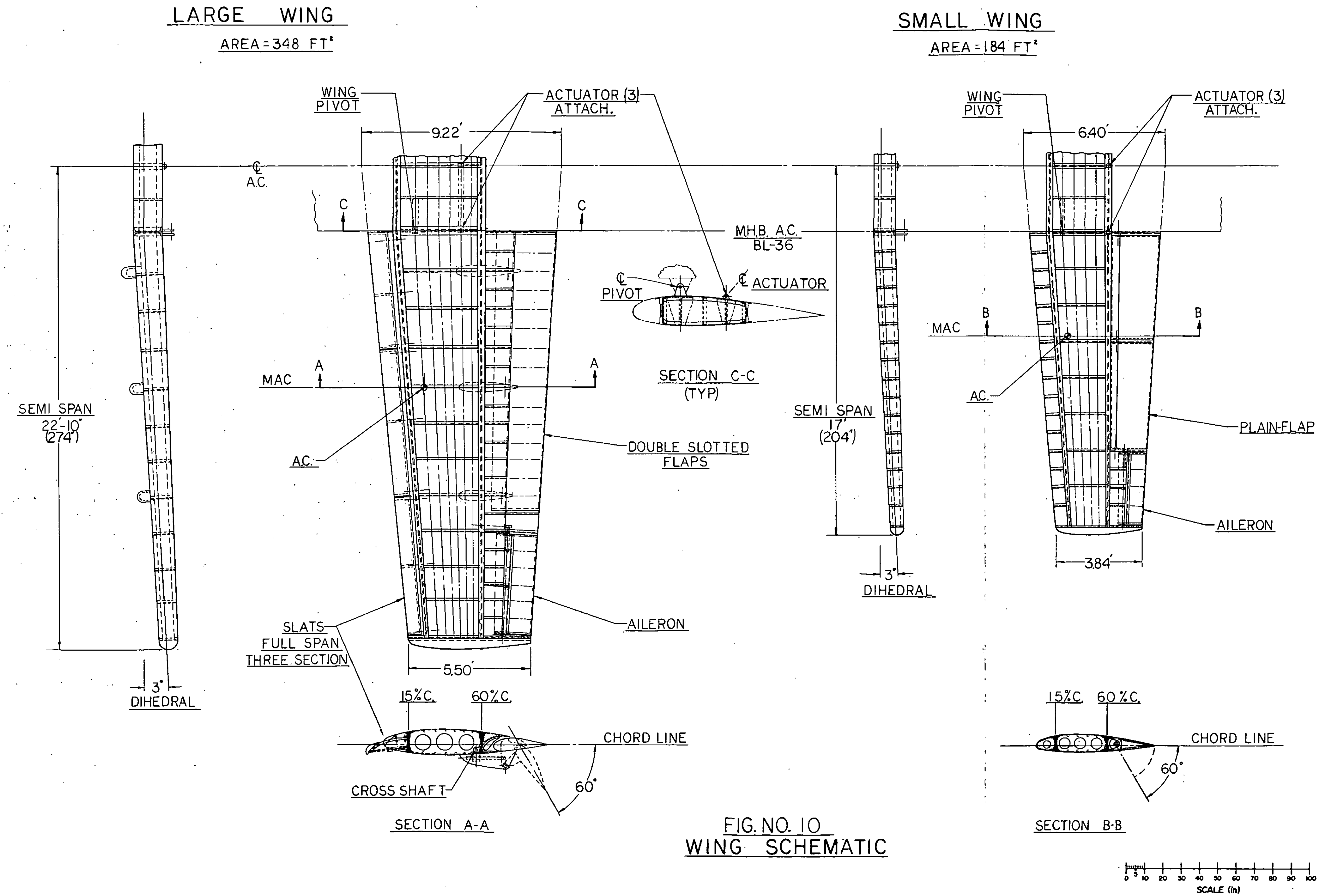


FIG. NO. 10
WING SCHEMATIC

Alighting Gear

The RSRA incorporates a main gear forward/tail wheel landing gear system. All wheels are fully retractable and are enclosed by sealed, aerodynamically smooth doors. The landing gear system is designed for landing speeds up to 120 knots with an 8 feet per second sink speed. The braking system is designed for eight feet per second squared deceleration at speeds up to 120 knots. This aircraft is designed to operate from prepared surfaces.

Main Wheels

The main landing gear, Figure 11, uses a conventional air-oil type oleo. It absorbs the landing energy of the aircraft and provides an air cushion for smooth taxi. The shock absorber is mounted so that tire travel during strut compression and extension is essentially vertical. This prevents tire roll over during landing. The strut consists of a forged aluminum outer housing and an inner chrome-plated steel piston. The air and oil are separated by a floating piston to prevent aeration of the oil and foaming of the fluid during servicing and operation. A tapered metering pin passes through the orifice during strut compression to improve oleo efficiency. The gear has an oleo stroke of ten inches. Single main wheels equipped with hydraulically-operated brakes are attached to the axle at the lower end of the piston. Torque arms, attached to the piston and cylinder to prevent swiveling, are hinged to permit vertical travel. A side strut connects the shock strut to a fitting near the center line of the fuselage on a bulkhead to react side loads. A shock strut to react fore and aft load and vertical load is mounted to the airframe across two bulkheads. The landing gears retract inboard on a 20 degree cant to bypass each other and minimize the system envelope. Hydraulic actuation is utilized to retract the landing gear. An uplock hook maintains the gear in the retracted position. Maintenance requirements are kept to a minimum by the use of nonlubricated spherical bearings. A standard air-charging chuck and gauge can be attached to introduce air or nitrogen under pressure into the strut. Tie down rings on the gear are used in conjunction with fuselage tie down points to secure the aircraft in high winds. Normal towing is accomplished from the tail gear axle by utilizing standard tow bars, as listed in MIL-STD-805.

Brakes

Brakes are provided to stop the aircraft on landing, to assist in steering, and for parking. Left and right brakes are actuated separately by the force on the toe pedals located on the rudder bar. Depressing the toe pedal actuates the master cylinder piston, causing hydraulic fluid under pressure to be transmitted through the parking brake valve and into the main wheel brake units. The fluid forces the brake pucks against the discs. Pulling up on the parking brake valve handle, with brakes applied, traps fluid under pressure at the wheel brakes for parking. To release the pilot depresses the toe brakes.

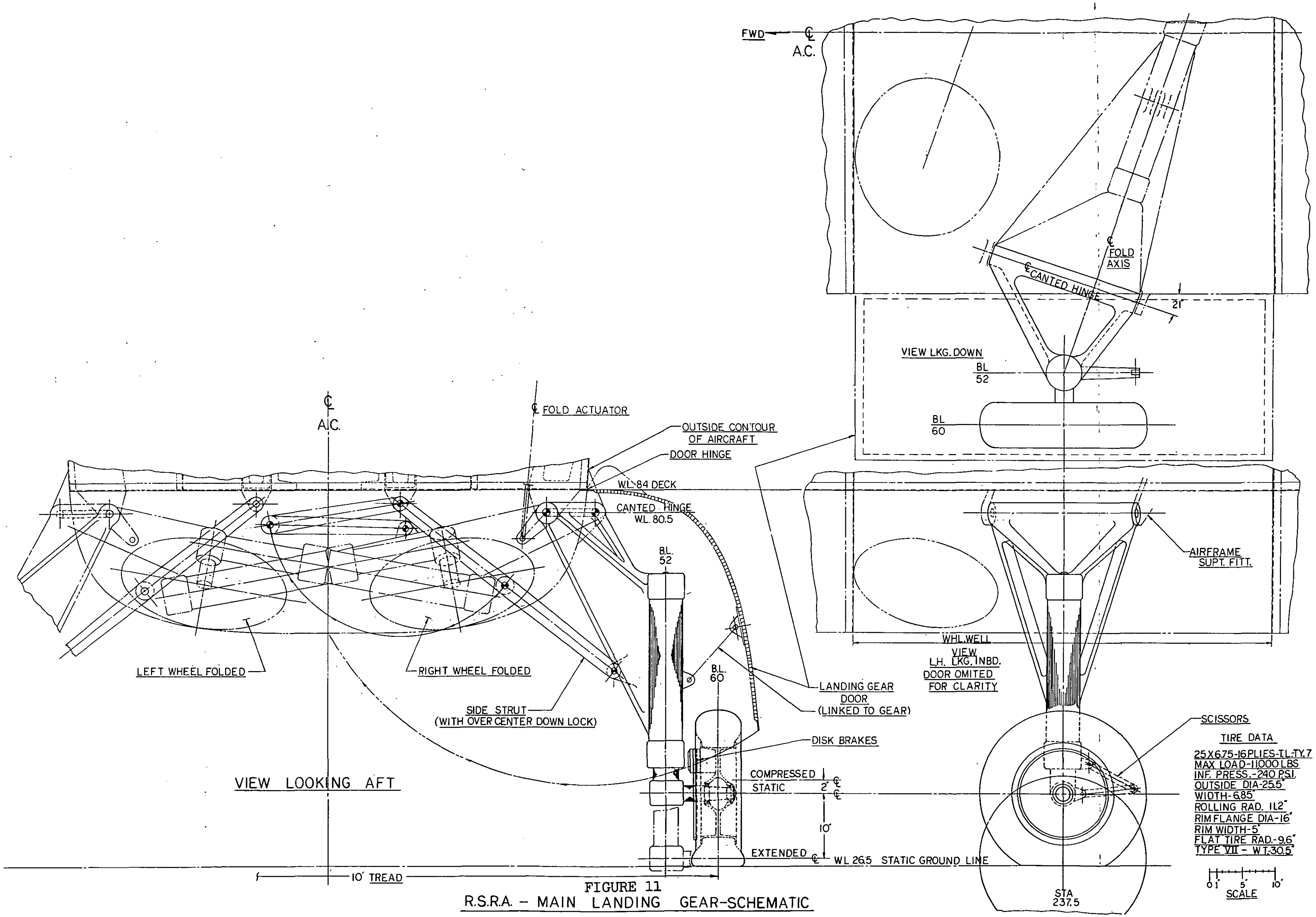


FIGURE 11
R.S.R.A. – MAIN LANDING GEAR-SCHEMATIC

Tail Gear

A 360 degree swiveling tail wheel is provided. The tail gear, Figure 12 is fully retractable and is housed in an aerodynamically smooth fairing when retracted. Five degrees of caster is incorporated to provide stability during ground operations. The tail wheel has a stroke of ten inches. A lock pin, electrically operated, keeps the wheel aligned with the flight path for landing or parking. An electrically operated switch, accessible to the pilot, controls the pin position. The drag strut engages an uplock hook to maintain the gear in the retracted position.

The tail gear consists of a shock strut support, outer housing, and a single wheel and tire supported by a fork and axle. The universal mounted shock strut consists of a housing and piston with separated air and oil chambers and a metered orifice to absorb energy. The wheel is a conventional split rim tubeless type to facilitate tire changing. It has tapered roller bearings, and seals are provided to keep out foreign matter.

A back up landing gear extension system, operable by either the pilot or copilot releases compresses nitrogen gas into the actuating cylinder, forcing the gear down and locked. The release sequence is electrically initiated.

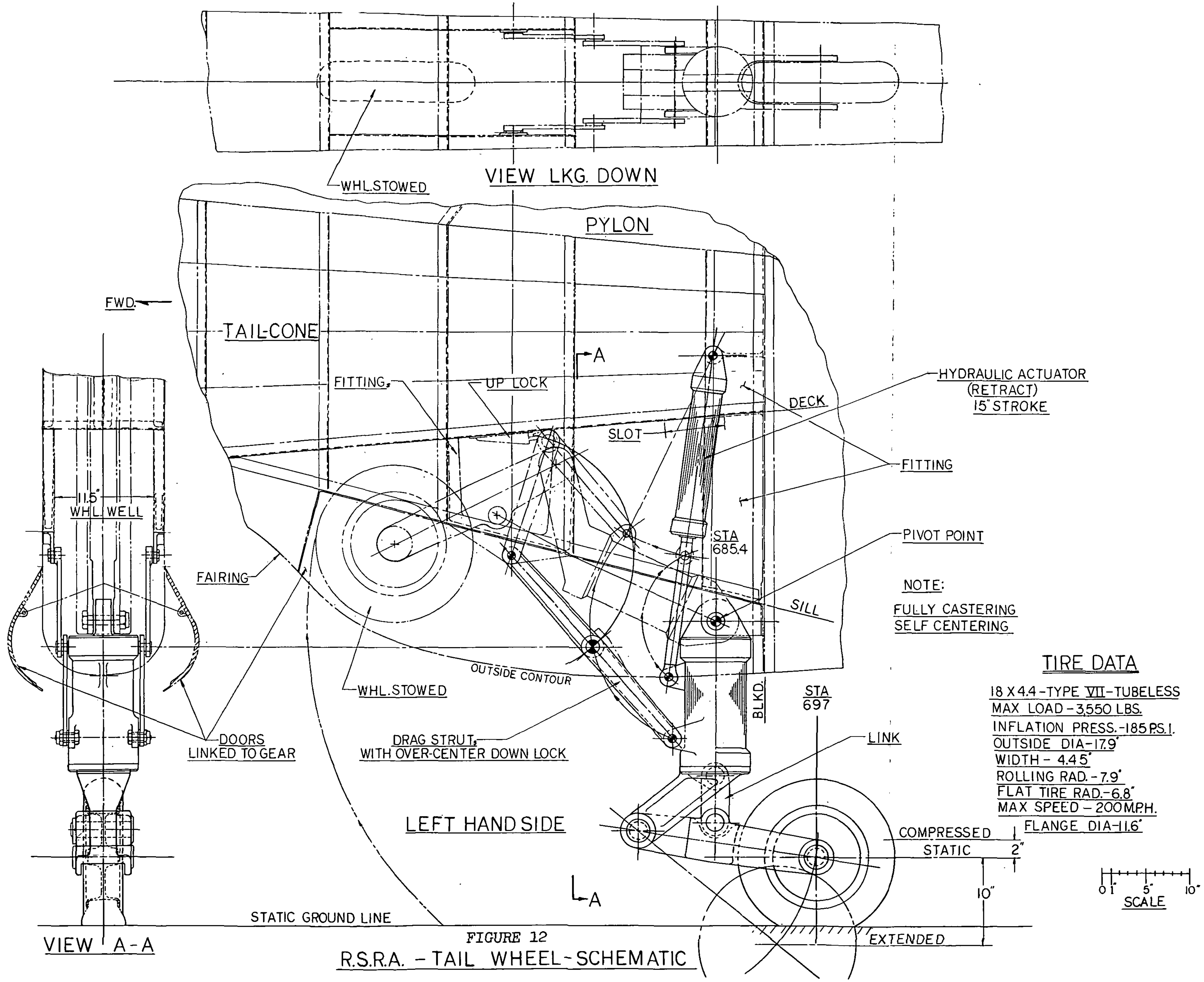


FIGURE 12
R.S.R.A. - TAIL WHEEL - SCHEMATIC

Special Onboard Data Systems

The special onboard data systems consist of the instrumentation which is new in concept and which has received little or no previous flight testing. The following systems are included in this category; (1) rotor force and moment measurement, (2) wing force and moment measurement, (3) auxiliary propulsion thrust measurement, and (4) anti-torque thrust measurement.

The purpose of the special onboard data systems is to measure the forces and moments produced by the main thrust and lift generating devices. In order to achieve these measurements isolation between the force or moment producing device and the aircraft fuselage must be made by the measuring transducer. In each case, this isolation is accomplished through the use of single or two axis load cells. The load cells in general do not measure the required force directly, but require resolving based on the geometry and particular force or moment under consideration.

Load cells are the logical choice for the special onboard measurements since they can be included in the original aircraft design and configured to isolate the loads as required. Accuracies of typical load cells are better than 1% of applied load with overload protection of two to three times full scale rating. Since the accuracy of the load cells are a function of applied load and the individual load cells measure the summation of several forces, the resulting accuracies are dependent upon geometry of the overall measuring system and placement of the individual load cells.

Description of the Measuring Systems

Rotor System. - Two rotor force measuring systems have been conceptually designed for the RSRA. The first uses load cells to measure all rotor forces and moments. The second replaces the load cells with hydropneumatic actuators to provide load sensing as well as active rotor vibration suppression. The system shown in Figure 13 illustrates the load cell measuring system. A detailed description of the force measurement system for the optional Active Vibration Suppression System is presented in a later section of this report.

The load cell rotor force measuring system uses four vertical and three horizontal load cells to measure all rotor forces and moments. These cover all load paths between the gearbox and the airframe, so that all loads can be measured. The load cells are mounted through spherical bearings so that only axial loads will be transferred through each load cell.

The load cell is Interface Inc., Scottsdale, Arizona, model 1030. It has a load measurement range of 50,000 pounds and a safe overload value of 150,000 pounds. It can withstand more than 10^8 fully reversed cycles without failure.

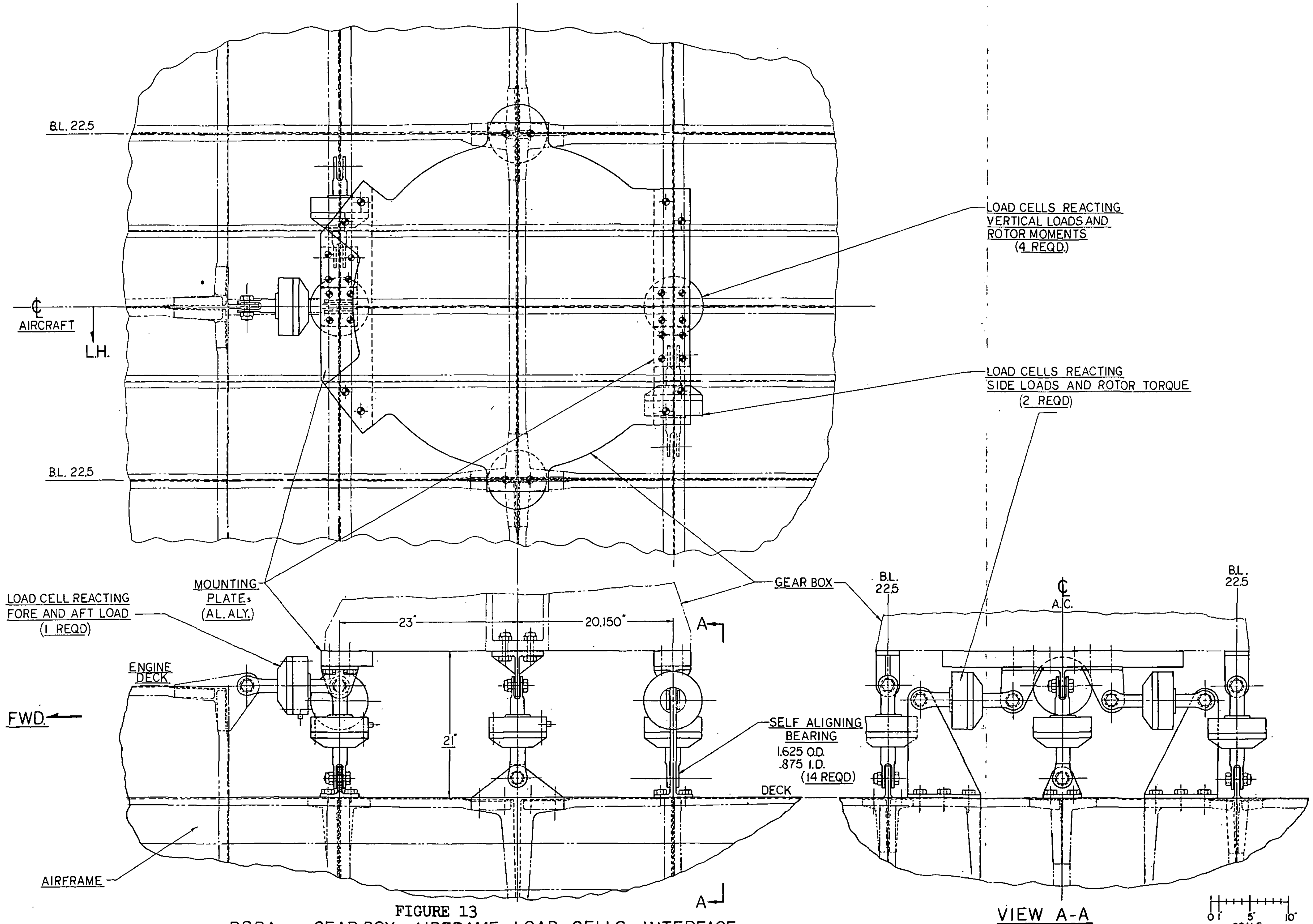


FIGURE 13

R.S.R.A. — GEAR BOX-AIRFRAME-LOAD CELLS-INTERFACE

Wing force measurement. - The wing force measurement system is shown in Figure 14 and consists of two 2-axis load cells at the 25% wing chord and three single-axis load cells in series with the three actuators. This configuration provides the wing force and moment measurement as discussed in Appendix A. The 2-axis load cells are available through Revere Electronic Division of Neptune Meter Co. Wallingford, Conn. The single-axis load cells are available from Interface Inc.

Auxiliary propulsion thrust measurement. - The auxiliary propulsion thrust measurement system is shown in Figure 15. A single-axis load cell mounted in-line with the cross member measures the desired thrust. Interface load cells can be used for this measurement.

Anti-torque thrust measurement. - The fan/duct assembly is a free floating unit in the direction of thrust. The assembly attaches to the aircraft frame by means of three temperature compensated load cells 120 degrees apart on the fan housing. The thrust system mount permits no torque component in the thrust measurement system and only sufficient axial movement for load cell deflection (typically 0.005 to 0.010 inch full scale). Outputs of the load cells sum to produce total fan thrust and the position of the center of thrust.

Accuracy Requirements

The required accuracy of any measurement system is dependent on the immediate mission or task to be performed. Each task usually will accept different measurement accuracies and no single accuracy can be attached to all tasks. For example, for automatic control work the quantities measured may require accuracies to 5% of the test condition, whereas in comparing performance of two rotor systems the required accuracies may be only 2% of test condition.

It is generally felt that an upper limit on acceptable accuracy in general might be 5% of the design load. At fractions of design load less accuracy can be tolerated.

Expected Accuracies for the Special Measurement Systems

Rotor system. - Accuracy studies were performed for two rotor system configurations. Configuration A is shown in Figure 16. Configuration B places the load cells as shown previously on Figure 13, page 33. Each transducer was assumed to have an accuracy of 1% of applied load ±30 lbs. The equations used and details of the accuracy analysis can be found in Appendix A.

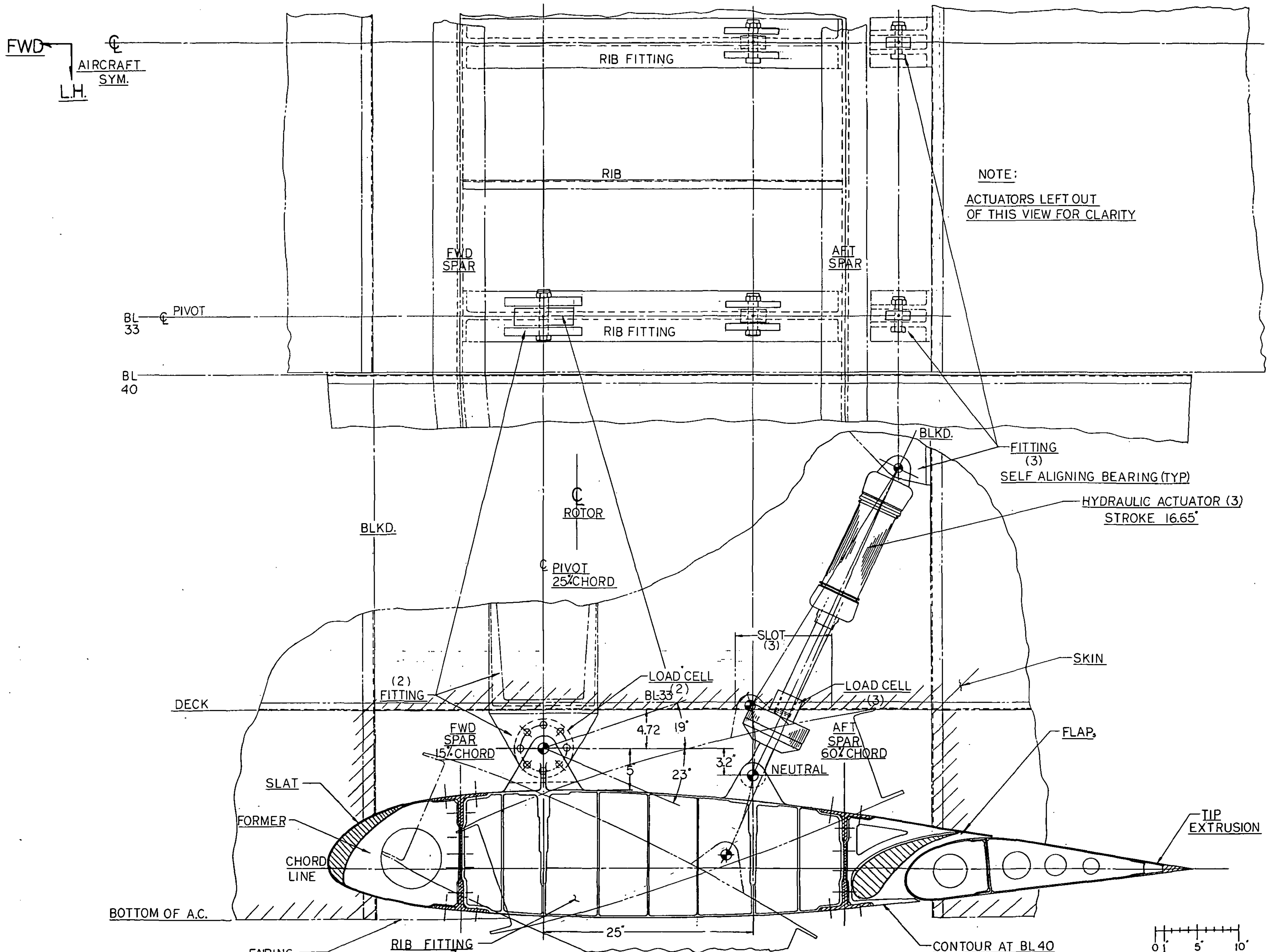
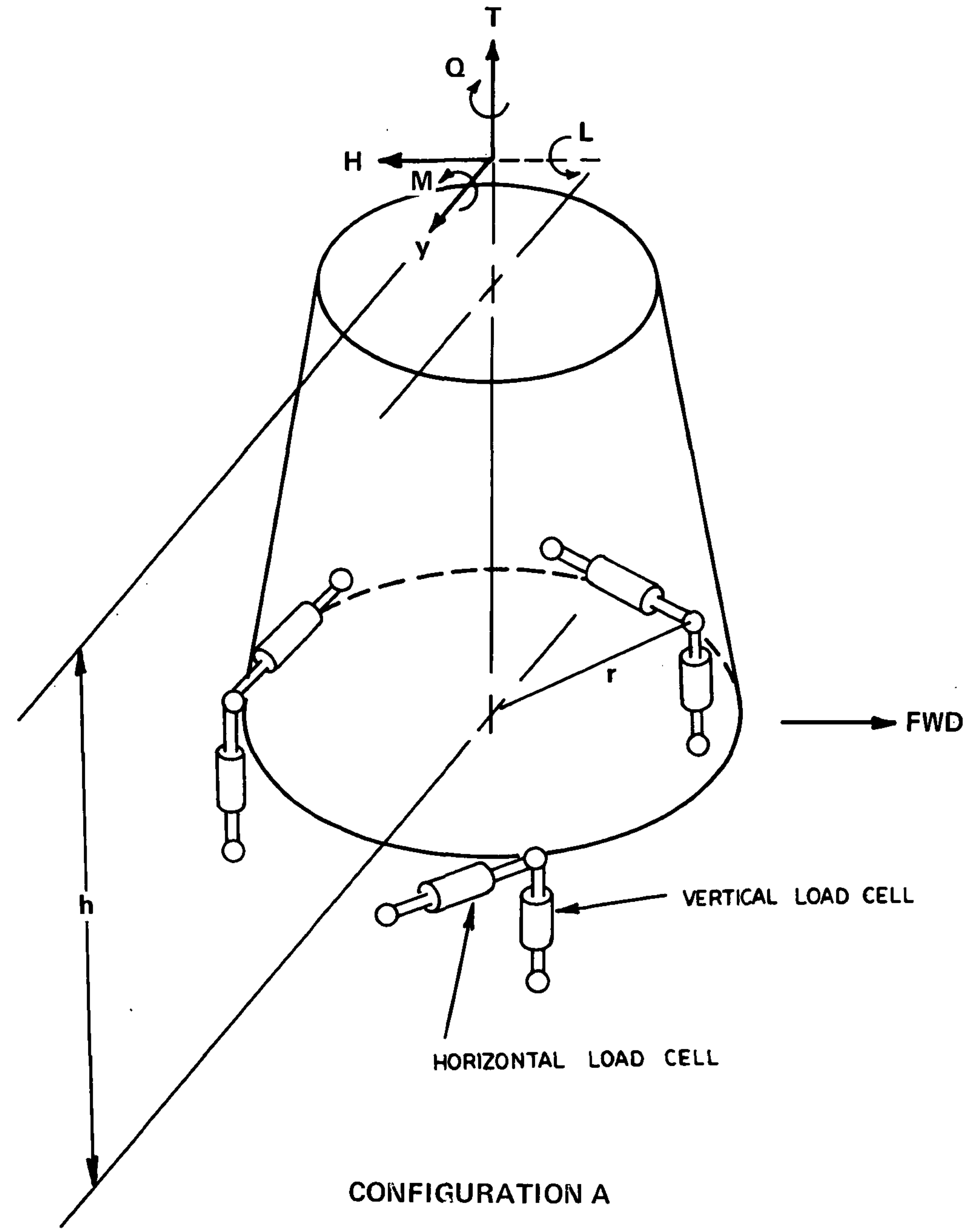


FIGURE 14 R.S.R.A.-WING TILT-AIRFRAME-INTERFACE



CONFIGURATION A

ROTOR FORCE MEASUREMENT SYSTEM

FIGURE 16

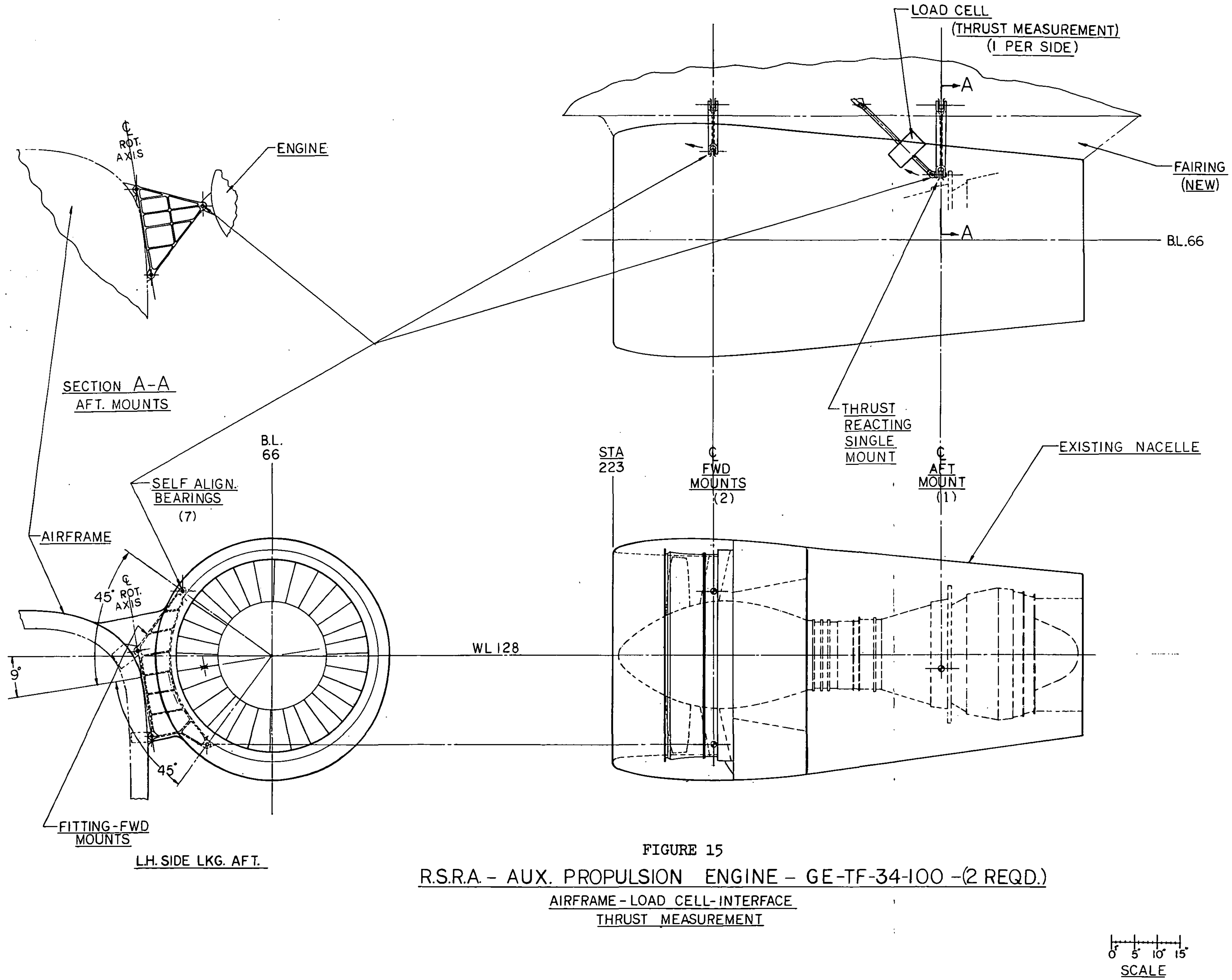


FIGURE 15

R.S.R.A. - AUX. PROPULSION ENGINE - GE-TF-34-100 -(2 REQD.)

AIRFRAME - LOAD CELL - INTERFACE
THRUST MEASUREMENT

Accuracy curves for Configuration A are shown in Figures 17 through 20. Figures (17) and (18) show typical accuracies for rotor torque and rotor thrust plotted against representative applied loads. The overall accuracy is seen to be better than 1%. Figures (19) and (20) show typical accuracies for rotor hub pitching moment and longitudinal force. The resulting accuracy is shown to be very dependent upon main rotor torque. This is due to the large forces developed in the horizontal transducers due to rotor torque. Since the transducer accuracy is based on 1% of applied load, the torque contribution in the horizontal transducer causes poor accuracy in the measurement of hub pitching moment and horizontal hub force. Hub rolling moment and lateral hub force accuracies are very similar to pitching and horizontal accuracies respectively. The test point condition as discussed in the RFP section 4.1.2c was analyzed using Configuration A. The results are presented in Table III. The same test point was analyzed for Configuration B and is also presented in Table III. The resulting reduction in uncertainty is clearly seen in the longitudinal force and pitching moment while an increase in the lateral force and rolling moment uncertainty resulted for Configuration B. This change is for the most part due to the torque contribution which now affects the lateral force and rolling moment only.

The configuration presented in Figure 13 provides for excellent accuracy in thrust, torque, pitching moment and longitudinal force ($\approx 1\%$) at a sacrifice in lateral force and rolling moment.

TABLE III
EFFECT OF CONFIGURATION
CHANGE ON ACCURACY
(ROTOR MEASUREMENT SYSTEM)

MAIN ROTOR HUB FORCES	TEST CONDITION	CONFIGURATION A (1σ ACCURACY)	CONFIGURATION B (1σ ACCURACY)
Long.	1380 lbs	± 185 lbs	± 33 lbs
Lat.	0 lbs	± 171 lbs	± 216 lbs
Thrust	18000 lbs	± 105 lbs	± 115 lbs
Roll M.	0 ft-lbs	± 1000 ft-lbs	± 1296 lbs
Pitch M.	6750 ft-lbs	± 1080 ft-lbs	± 300 lbs
Torque	60000 ft-lbs	± 416 ft-lbs	± 432 lbs

ROTOR FORCE MEASUREMENT UNCERTAINTY

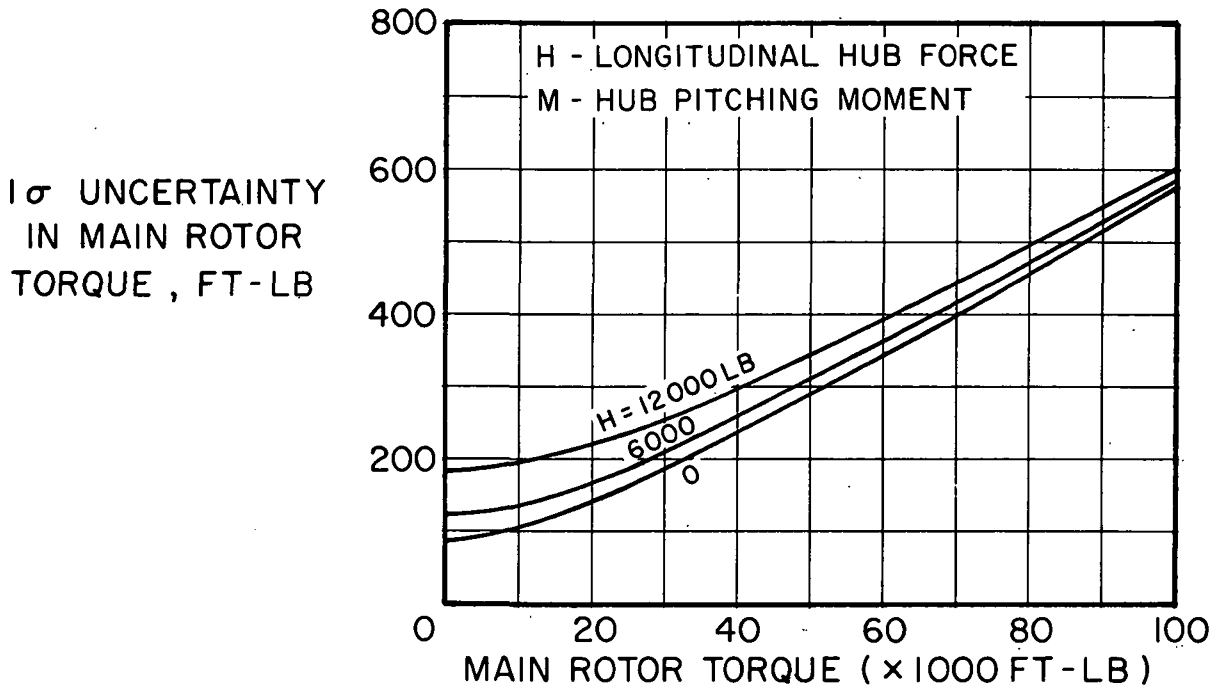


FIGURE 17

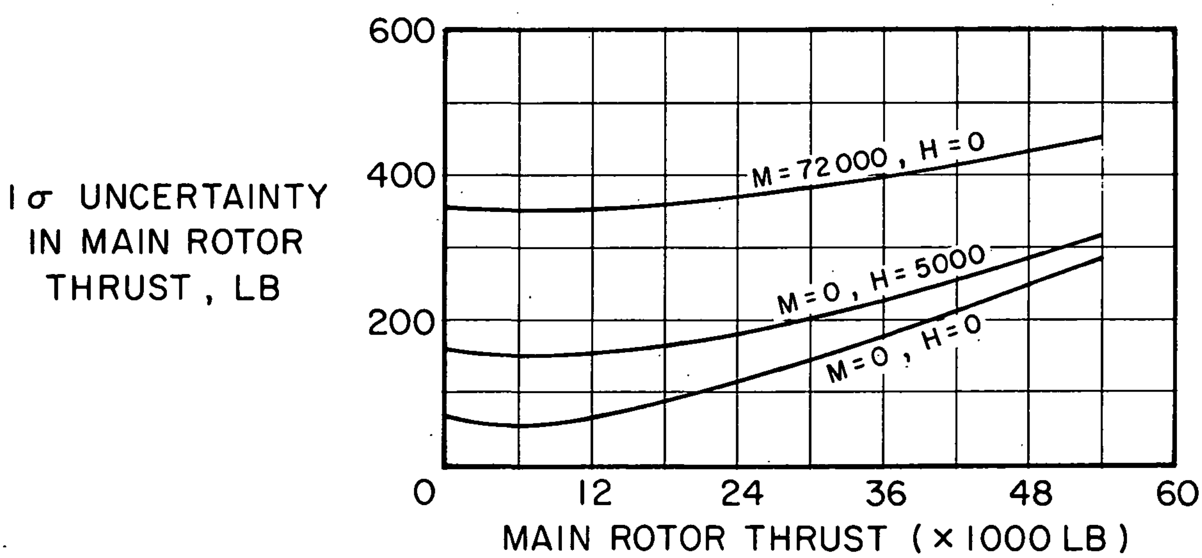


FIGURE 18

ROTOR FORCE MEASUREMENT UNCERTAINTY

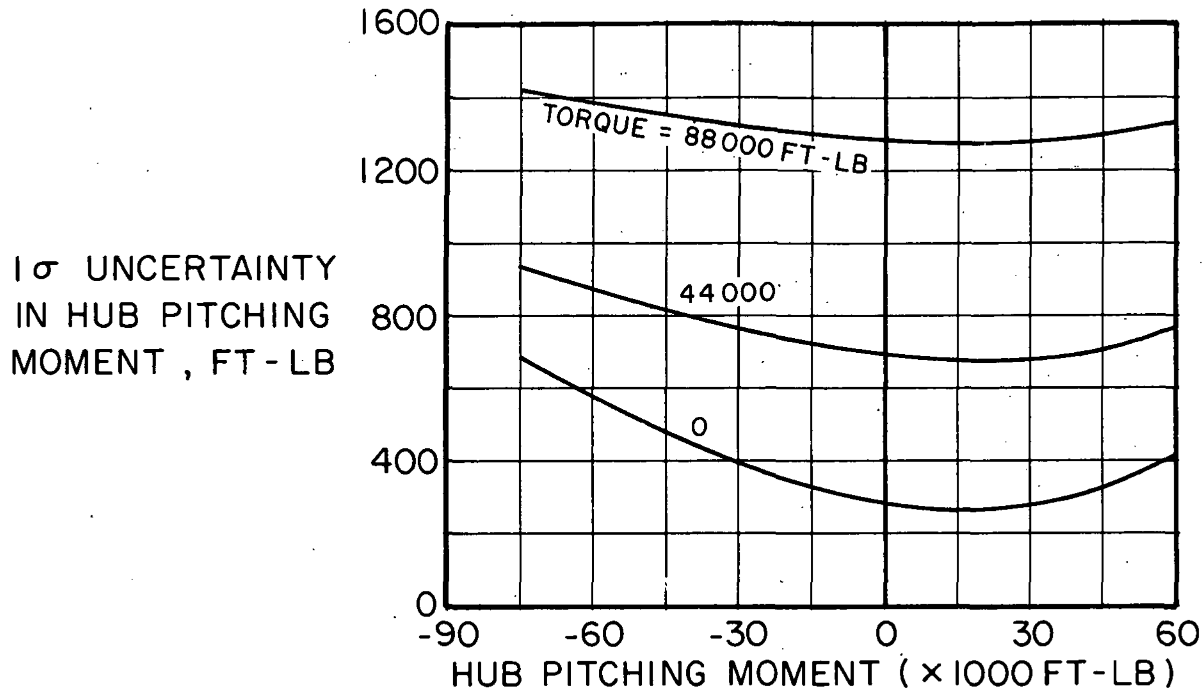


FIGURE 19

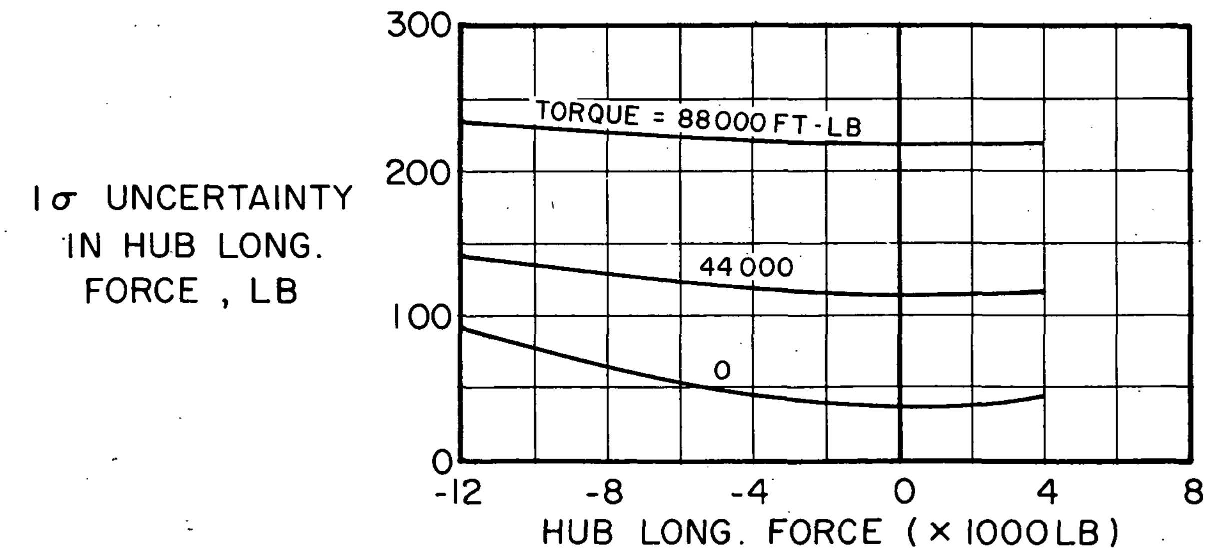


FIGURE 20

Wing system. - The wing accuracy study was performed in a manner similar to the rotor system. The details and equations were developed similar to that of the rotor. Figure 14 shows the wing diagram which was used in the accuracy study. The wing accuracy study was performed by writing the force and moment equation for this configuration. The transducers were assumed to be accurate to 1% of applied load. The resulting accuracy equations showed that worst accuracy is obtained when wing lift and pitching moment are greatest and the wing angle of attack is large. Table IV shows the resulting accuracy for V = 300 knots and is seen to be better than 1%. The resulting accuracy for worst case is shown in Table V and is better than 2%. These results indicate that this wing measurement system concept can provide good accuracies. Good alignment and calibration must be made in order to achieve these accuracies.

The test point as discussed in the RFP was applied to the wing. At this condition, good accuracies are obtained and are shown in Table VI.

Auxiliary propulsion. - The main consideration to accuracy with the auxiliary propulsion thrust measurement is the ability of the load cell to have a high resistance to extraneous forces. Flat load cell designs maintain accuracy to 0.1% for extraneous forces up to 100% of full rating. An accuracyof better than 2% is expected for auxiliary propulsion thrust measurement.

Anti-torque. - The anti-torque system should also maintain accuracy of better than 2% for thrust. Torque measurement can be accomplished by a strain gauged shaft and yield accuracies of about 2% of full scale.

TABLE IV
WING ACCURACY RESULTS

CASE 1

V = 300 Knots
Full Wing Loading
$\alpha = 3^{\circ}$

WING FORCE	TEST CONDITION	ACCURACY 1σ
Lift	25000 lbs	± 235 lbs
Drag	2000 lbs	± 17 lbs
Pitch M.	15000 ft-lbs	± 150 ft-lbs
Roll M.	0 ft-lbs	± 448 ft-lbs
Yaw M.	0 ft-lbs	± 34 ft-lbs

TABLE V

CASE 2 (Worst Case)

V = 120 Knots
Full Wing Loading
$\alpha = 13^{\circ}$, Flaps Down

WING FORCE	TEST CONDITION	ACCURACY 1σ
Lift	25000 lbs	± 405 lbs
Drag	7600 lbs	± 143 lbs
Pitch M.	75000 ft-lbs	± 750 ft-lbs
Roll M.	0 ft-lbs	± 870 ft-lbs
Yaw M.	0 ft-lbs	± 230 ft-lbs

TABLE VI

CASE 3 (Test Point)

V = 150 Knots
Wing Loading = .4 GW
$\alpha = 5^{\circ}$

WING FORCE	TEST CONDITION	ACCURACY 1σ
Lift	12000 lbs	± 42 lbs
Drag	1040 lbs	± 4 lbs
Pitch M.	5500 ft-lbs	± 55 ft-lbs
Roll M.	0 ft-lbs	±268 ft-lbs
Yaw M.	0 ft-lbs	± 27 ft-lbs

Calibrating the Onboard Data Systems

Hover thrust. - In addition to the measurements described above, the RSRA is configured to measure hovering thrust by tethering the helicopter to a "dead man" from the center of gravity of the aircraft. Various lengths of cables will be used to attain different rotor heights above the ground to allow in and out of ground effect testing.

The measurement system will consist of a tension load cell in series with the tether cable and two pendulums located 90° apart attached to the load cell to indicate the vertical orientation of the cable. The pilot will have visual indications of cable tension, as well as longitudinal and vertical cable angle. There will also be analog electrical outputs for recording the same information.

Calibration of the tethering system load cell will be done in the structural calibration laboratory using a calibration "standard" load cell in series with it. The muscle for applying the load will be either a tensile machine or hydraulic strut. The longitudinal and lateral pendulums will be calibrated using a bubble protractor as a standard.

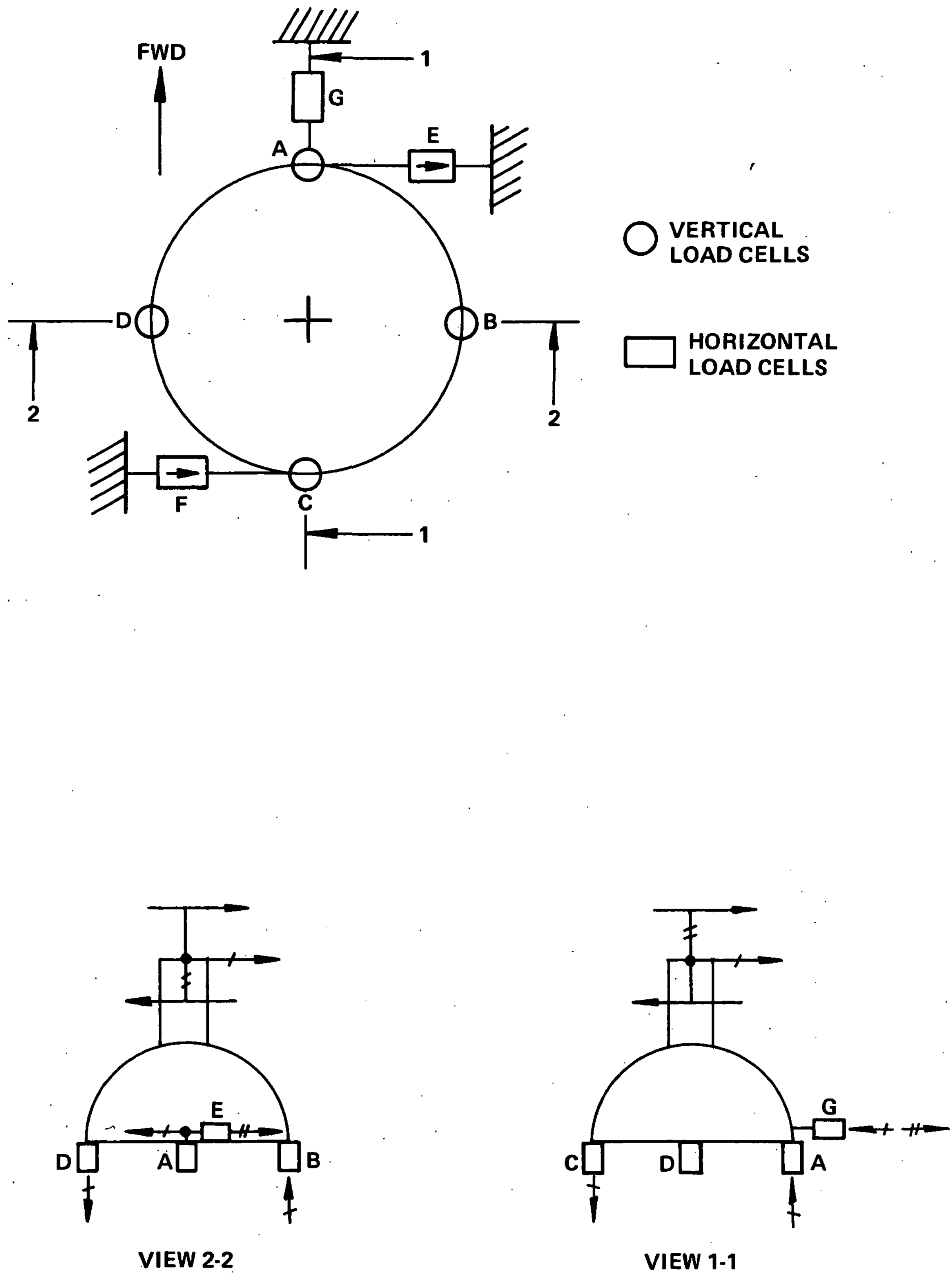
Rotor forces and moments. - Figure 21 is a schematic showing the placement of the load cells which react all of the forces that are applied by the main rotor.

The cells which react the various forces are as follows:

(1) Torque - cells E & F
(2) Thrust - cells A, B, C, & D
(3) Long, Shear - cell G, View 1-1
(4) Long Moment - cells A, G, C View 1-1
(5) Lat. Shear - cells E, F View 2-2
(6) Lat. Moment - cells B, D, E, F View 2-2

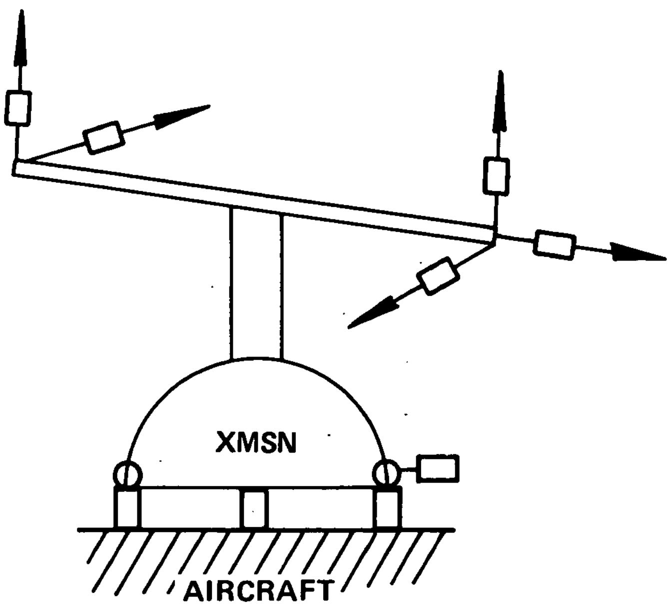
The procedure for the calibration of this system will be to apply incremental loads of each of the inputs individually. Following this, loads in various combinations will be applied to quantify the cross talk effects. Combined loads investigations will be concentrated in the area of calculated operational loads. The outputs from all transducers will be recorded during the application of all the input loads.

The various loads will be applied with the use of hydraulic struts and "standard" load cells attached to a beam installed in place of the main rotor head, as schematically shown in Figure 22. This calibration shall be done as a system calibration, with the complete aircraft, to have all the deflections present at the time of calibration.



ROTOR FORCE MEASUREMENTS

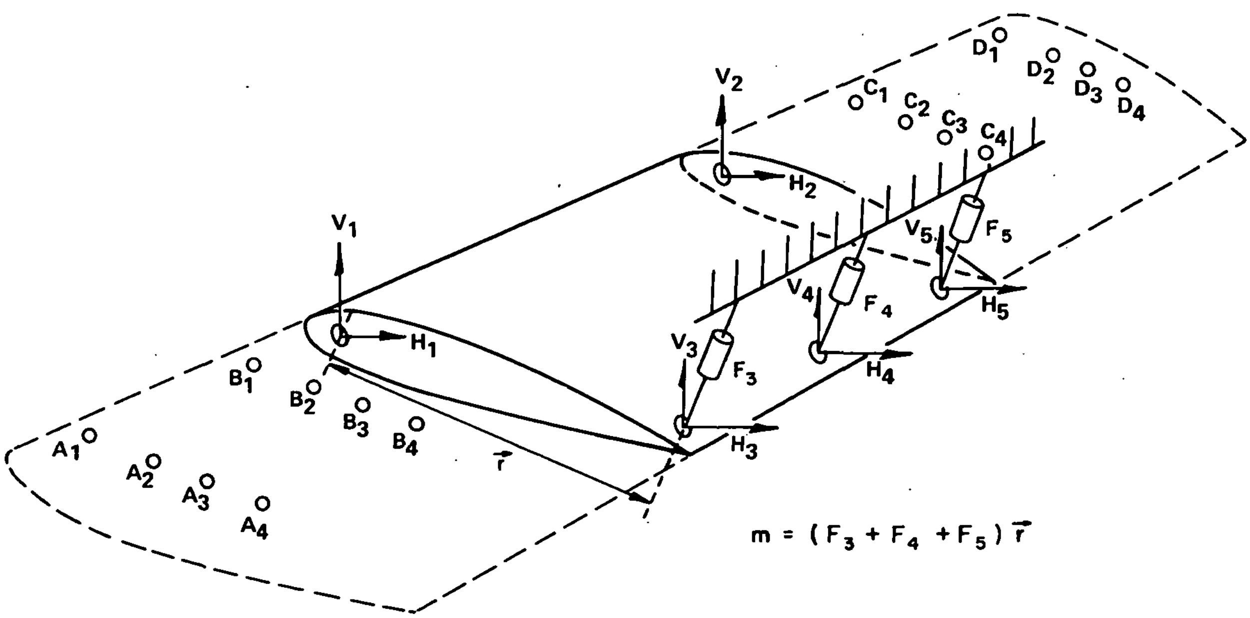
FIGURE 21



ROTOR CALIBRATION FIXTURE

FIGURE 22

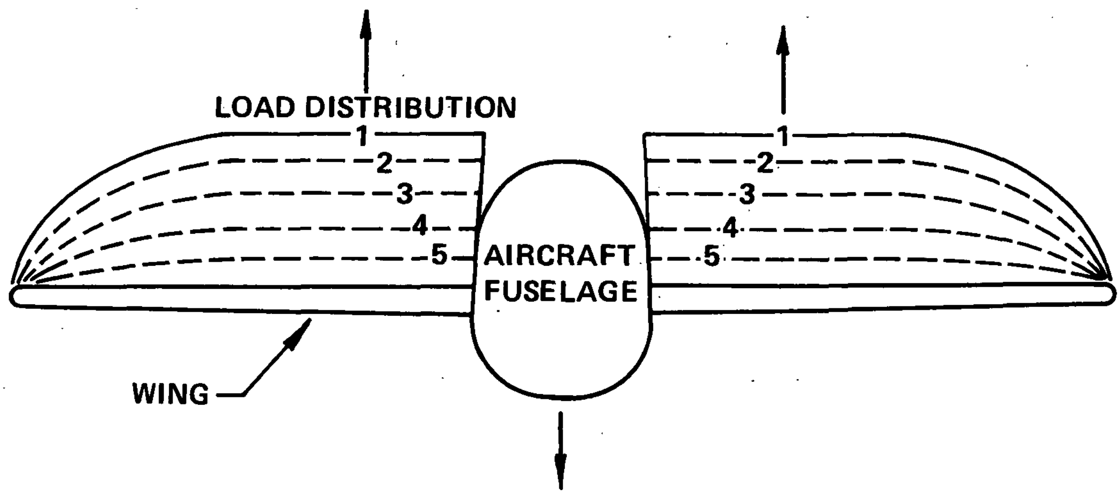
Wing forces and moments. - Wing lift is the sum of all of the vertical forces and drag is the sum of all the horizontal forces, as shown in Figure 23.



WING FORCE MEASUREMENT

FIGURE 23

The principal calibration of the wing lift measurement will be to apply several increments of load distributed spanwise along the wings as shown in Figure 24.



WING LOAD DISTRIBUTION

FIGURE 24

These loads will be distributed chordwise about the calculated normal chordwise center of lift. This calibration will be repeated at several angles of wing incidence.

Other tests will be performed to evaluate the lift measurement system as follows: (Refer to Figure 23 for location of loads)

(1) Equal drag loads will be applied with no lift loads to quantify any "cross talk" that may be present.

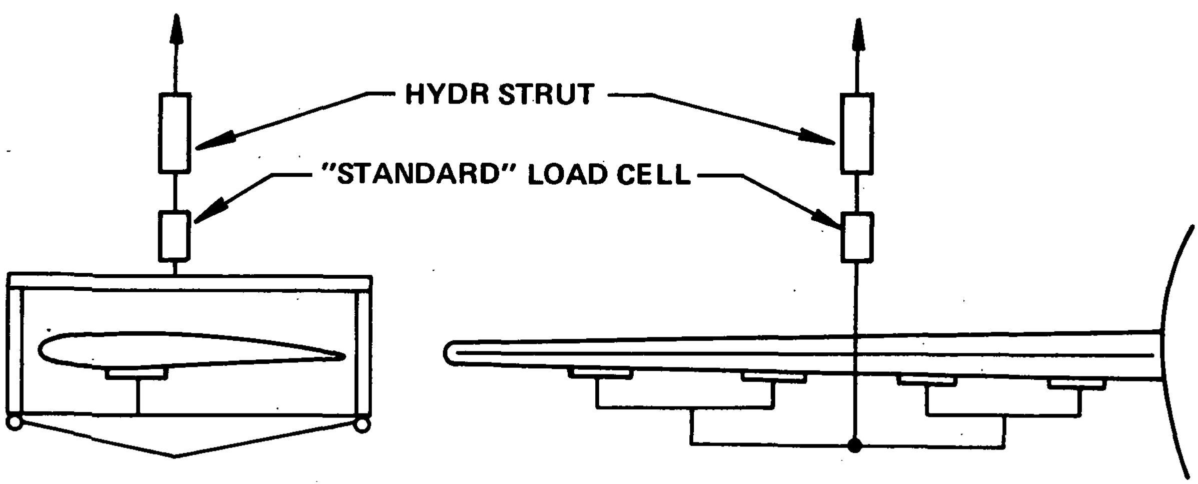
(2) Unequal drag loads will be applied with no lift loads for the same purpose.

(3) The same concentrated vertical loads will be applied in pairs at A_2 and D_2, A_4, B_2 and C_2, and B_4 and C_4 to quantify the capability of the lift measurement system to measure vertical loads regardless of the applied location.

(4) The same single vertical load will be applied at A_2, A_4, D_2 and D_4 for the same purpose.

(5) The same concentrated vertical loads will be applied in pairs as in (3) above in the presence of the calculated maximum drag load applied to the wings, to evaluate combined loading effects.

Application of lift loads will be as shown in Figure 25 through a whittle tree arrangement using a "standard" load cell and a hydraulic strut for the "muscle."



WING LOAD APPLICATION

FIGURE 25

The principal calibration of the wing drag measurement will be to apply several increments of drag load, up to the maximum calculated, to the leading edge of the wings. The calibration will be repeated at the same angles of incidence as used during the lift calibration.

Other tests will be performed to evaluate the drag measurement system as follows:

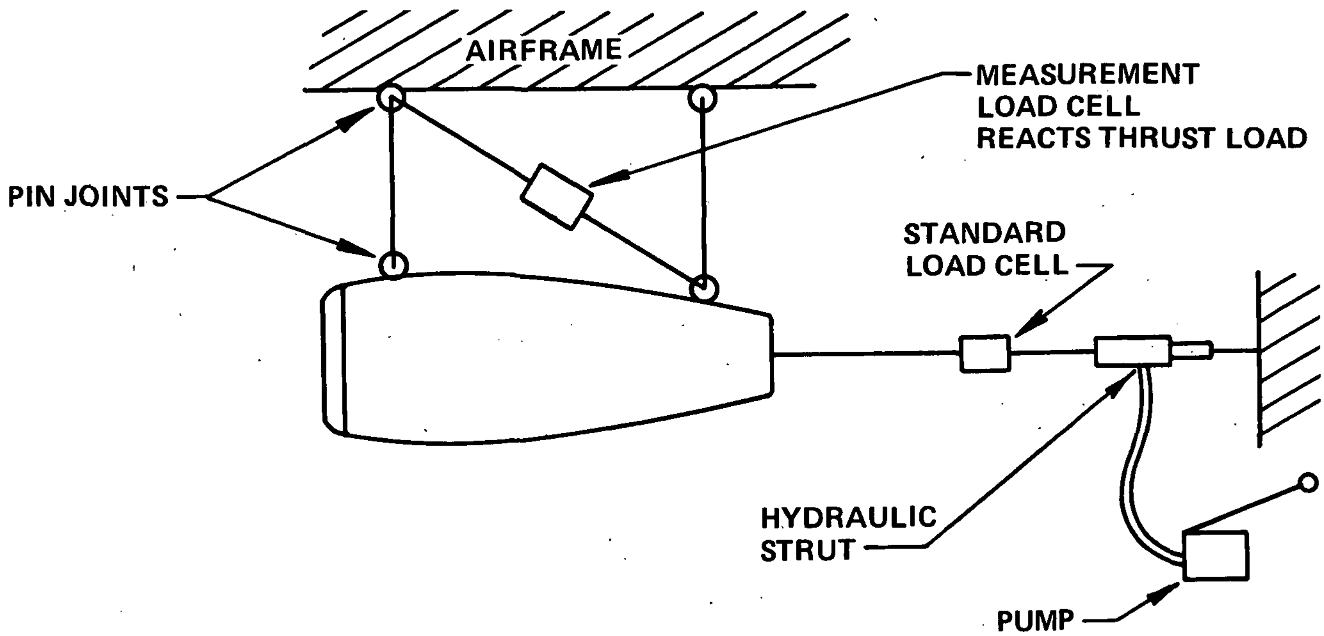
(1) The same single drag load will be applied at points A_1, B_1, C_1 and D_1 to quantify the capability of the drag measurement system to measure drag loads regardless of the distribution.

(2) Maximum calculated drag loads will be applied in the presence of the calculated maximum distributed lift loads, to evaluate combined loading effects.

The output from all transducers will be read during the application of all the various combinations of lift and drag loads.

Drag loads will be applied in the same manner as lift loads except in the horizontal plane.

Auxiliary propulsion. - Calibration of the auxiliary propulsion measurement will be achieved as shown schematically in Figure 26. The measurement load cell will be compared to a "standard" load cell that is used to monitor the calibration force simulating engine thrust. The muscle for load application will be a hydraulic strut of suitable size.



AUXILIARY THRUST MEASUREMENT

FIGURE 26

Anti-torque thrust. - Calibration of the anti-torque thrust measuring system will be similar to that described above for the auxiliary propulsion system. A fixture is required to distribute the test load to all three load cells.

Additional Aircraft Data Acquisition Equipment

Table VII is a listing of the additional measurement capability that is to be provided on the aircraft in addition to the load cells. Appropriate transducers and signal conditioning are to be installed and wired to a magnetic tape recording system. The magnetic tape system shall consist of:

- 1 inch, 14 track, IRIG magnetic tape recorder with intermediate bandwidth record electronics

- Time code generator

- PCM multiplexer/encoder with control unit

- Proportional bandwidth voltage controlled oscillators (IRIG F.M. bands 7-16, 10 dynamic measurements per tape track) with summing amplifiers and reference oscillators

- Master tape control unit

- Control track reference oscillator

In order to achieve data system compatibility between NASA Langley and contractor equipment, the following recommendations are proposed:

- The Contractor shall purchase a time code generator (NASA 36 BIT TIME CODE) for the testing at NASA Langley. During testing at the contractor's plant, the contractor shall use a contractor-owned time code generator (modified IRIG B Code).

- Since the proportional bandwidth F.M. system has been selected as the most effective configuration to record the aircraft's dynamic data, the contractor suggests that NASA obtain the required discriminatory tuning units/filters to complement its constant bandwidth F.M. system with a proportional bandwidth F. M. system. This procurement will provide data system compatibility and added flexibility to the NASA ground station.

The contractor shall provide technical support to NASA in defining the magnetic tape layouts, establishing calibration techniques and engineering units conversion factors, deriving the PCM formats, selecting data sampling rates and filtering, familiarization training with the data system for NASA personnel, and reviewing system documentation.

TABLE VII
MEASUREMENT LIST

ITEM	LOCATION	DESCRIPTION	TOTAL NUMBER
1.	Engine-TF34	Gas Generator Speed	2
2.	Engine-TF34	Power Turbine Speed	2
3.	Engine-TF34	T5 Temperature	2
4.	Engine-TF34	Power Lever Position	2
5.	Engine-TF34	Fuel Flow	2
6.	Engine-TF34	Engine Vibration	16
7.	Engine-TF34	Engine Temperature	16
		Sub-Total	42
8.	Engine T-58-16	Gas Generator Speed	2
9.	Engine T-58-16	Power Turbine Speed	2
10.	Engine T-58-16	T5 Temperature	2
11.	Engine T-58-16	Power Lever Position	2
12.	Engine T-58-16	Fuel Flow	2
13.	Engine T-58-16	Engine Vibration	16
14.	Engine T-58-16	Engine Temperature	16
15.	Engine T-58-16	Torque	2
		Sub-Total	44
16.	Airframe	Total Stress Transmission Area	12
17.	Airframe	Total Stress Tail Cone	20
18.	Airframe	Total Stress Tail Pylon	12
19.	Airframe	Total Stress Stabilizer	12
20.	Airframe	Total Stress Wing	20
21.	Airframe	Total Stress Flap	12
22.	Airframe	Total Stress Ailerons	12
23.	Airframe	Total Stress Drag Brakes	12
		Sub-Total	112
24.	Main Rotor Head	Pitch Angle	1
25.	Main Rotor Head	Flapping Angle	1
26.	Main Rotor Head	Lag Angle	1
27.	Main Rotor Head	Push Rod Load	1
28.	Main Rotor Head	Stationary Star Load	3
29.	Main Rotor Head	Shaft Bending Upper	1
30.	Main Rotor Head	Shaft Bending Lower	1
		Sub-Total	9
31.	Main Blade Stresses	Edgewise Total Stress	8
32.	Main Blade Stresses	Back Radius Total Stress	8
33.	Main Blade Stresses	Flatwise Bending Stress	8
34.	Main Blade Stresses	Damper Load	1
		Sub-Total	25

TABLE VII (Cont'd)

ITEM	LOCATION	DESCRIPTION	TOTAL NUMBER
35.	Aircraft	Pitch, Roll, Yaw Attitude	3
36.	Aircraft	Pitch, Roll, Yaw Rate	3
37.	Aircraft	Pitch, Roll, Yaw Acceleration	3
38.	Aircraft	Indicated Airspeed	1
39.	Aircraft	Altitude	1
40.	Aircraft	Outside Air Temperature	1
41.	Aircraft	Rate of Climb	1
		Sub-Total	13
42.	Landing Gear	Vertical, Lateral, Drag Loads	6
43.	Landing Gear	Velocity	2
		Sub-Total	8
44.	Aircraft Vibration	Cockpit-Pilot	3
45.	Aircraft Vibration	Cockpit-Copilot	3
46.	Aircraft Vibration	Cockpit-Instrumentation Panel	3
47.	Aircraft Vibration	Center of Gravity	3
48.	Aircraft Vibration	Tail Cone	3
49.	Aircraft Vibration	Tail Pylon	3
50.	Aircraft Vibration	Wing	8
51.	Aircraft Vibration	Stabilizer	8
52.	Aircraft Vibration	Aileron	4
53.	Aircraft Vibration	Flaps	4
54.	Aircraft Vibration	Dive Brakes	4
		Sub-Total	46
55.	Stabilizer	Incidence Angle	1
56.	Stabilizer	Lift	1
57.	Stabilizer	Drag Load	1
		Sub-Total	3
58.	Wing	Flap Position	1
59.	Wing	Slat Position	1
60.	Wing	Wing Incidence	1
61.	Wing	Aileron Incidence	1
		Sub-Total	4
62.	Cockpit	Coll., Lat., Long., and Directional Control Position	4
63.	Cockpit	Horiz. Tail Gain Position	1
64.	Cockpit	Aileron Gain Position	1
65.	Cockpit	Rudder Gain Position	1
66.	Cockpit	Drag Brake Position	1
67.	Cockpit	Louver Position	1
68.	Cockpit	Thrust Position	1
			10

TABLE VII (Cont'd)

ITEM	LOCATION	DESCRIPTION	TOTAL NUMBER
69.	Computer	TF-34 Thrust Control Output	1
70.	Computer	Aileron Control Output	1
71.	Computer	Horizontal Control Output	1
		Sub-Total	3
72.	Fan-in-Fin	Hub & Blade Stresses	6
73.	Fan-in-Fin	Control Loads	2
74.	Fan-in-Fin	Shaft Bending	1
75.	Fan-in-Fin	Torque	1
76.	Fan-in-Fin	Control Position	1
77.	Fan-in-Fin	Door Position	1
		Sub-Total	12

Total Number of Additional Measurements: 331

Rotor Propulsion System

Selection of the T58-16 Shaft Horsepower Engines

Maximum rotor power requirements of the RSRA are determined by the hovering missions of the aircraft. For this case, the auxiliary thrust engines and the wing are removed. The gross weights for these missions are 20,276 lbs. This gross weight is within the capability of the S-67 rotor system chosen for the RSRA. The current engines installed in the S-67 are the T58-GE-5 engines. However, these engines do not have enough power for the RSRA missions. A survey was conducted to find an alternate engine. The results of the survey are shown below.

POWER ENGINES	SEA LEVEL STD STATIC MILITARY HORSEPOWER
T64 S4D-1	4110
T64-GE-413	3695
T64-S4D-1H	4400
T55-L-11	3400
LTC4V-1 (HLH)	7000
T58-GE-5	1400
T58-GE-16	1870
T53-L-13	1400
JFTD12A	4500
PT6T-4	1800

Most of the available engines were ruled out as being too large for the additional power requirement above the T58-GE-5 rating of 1400 HP. The obvious choice was to select the T58-GE-16, the growth version of the T58-GE-5, with 1870 HP. It has enough power to allow the RSRA to meet its hovering performance goals and to give the aircraft sufficient excess installed power to test rotors of higher disc loadings or lower efficiency than the baseline S-67 rotor. It also is the engine which the 3700 horsepower roller gearbox is designed for.

The T58-GE-16 engine is a gas turbine engine featuring a 10-stage axial flow compressor with a nominal pressure ratio of 8.4 to 1.0, a through-flow annular combustion chamber, a 2-stage axial-flow, aircooled gas-generator turbine and a 2-stage, rear drive free power turbine. The control system is an integrated hydromechanical and electrical system that provides isochronous power-turbine speed at any preselected RPM, independent of load within the specified power limitations of the engine. The control system has provisions for electrical interconnection of engines in dual-engine applications for load sharing. The control system automatically prevents overspeed of the gas generator or power turbine, overtemperature, compressor stall and combustion blow-out.

The engine is composed of two basic assemblies: gas generator assembly and the power turbine assembly. Power is taken off directly from the rear drive power-turbine output shaft at power turbine speed.

An oil-to-fuel heat exchanger is supplied with the engine and no additional heat exchangers are required for satisfactory operation of the engine lubricating system throughout the operating envelope. An integral engine-washing spray system is provided for washing deposits from the engine.

The ratings presented below are based on operation with no loading of the accessory drives, with no compressor bleed except that required for engine operation, no air bleed for anti-icing purposes and 100% ram recovery at the engine inlet.

TABLE VIII
T58-GE-16
PERFORMANCE RATINGS AT STANDARD, SEA LEVEL, STATIC CONDITIONS

RATING	MIN SHP	MAX SFC	MAX GAS GEN SPEED	RATED POWER TURBINE SPEED	MAX MEASURED POWER TURBINE INLET TEMP. (T_5)
Military	1870	.530	26,800	20,280	805°C (1481°F)
Normal	1770	.540	--	20,280	785°C (1445°F)
90% Normal	1593	.555	--	20,280	--
75% Normal	1328	.590	--	20,280	--
Ground Idle	--	180 lb/hr	15,400	0	--

The T58-GE-16 engine inlet is sized to minimize high speed inlet drag and to be within the engine air inlet distortion limits throughout the RSRA flight envelope. With the Inlet for the T58-GE-16 designed to this criteria, the power loss due to inlet pressure drop is estimated at 3 percent shaft horsepower. The exhaust pressure drop is considered to be negligible based on Sikorsky experience with similar T58 installations.

The RSRA operational requirements result in a variable RPM capability requirement for the T58 engines. In high speed forward flight with the wing providing the lift, the rotor must be slowed, yet supplied with enough power to retain control. This requires the T58 engines to successfully operate at reduced RPM's. For slowed rotors these can be as low as 40 percent of hover RPM. The T58-GE-16 engine is capable of this reduced RPM operation through modification of the control system. For the RSRA, one of the two T58 engines will be modified to operate at reduced RPM while the other engine will be a standard T58-GE-16 and will be reduced to ground idle for high speed, reduced RPM flight. Modification of only one of the two T58 engines reduces cost and maximizes reliability since one engine remains a standard T58-GE-16. Use of one engine to maintain the rotor at reduced RPM is feasible due to the low rotor power requirements in high speed flight.

This extensive power turbine speed range capability is not currently available with the T-58 engines. Discussions between Sikorsky and General Electric have revealed that such a feature could be provided. It will require a minor development effort and the cost for this development is included in the total RSRA cost estimates.

Auxiliary Propulsion System

The requirement which sizes the auxiliary thrust engines is the 300 knot test condition. This condition has the rotor with zero or a very light loading and the wing producing most of the lift. The drag sources are the rotor drag, the aircraft parasite drags, and the wing and rotor induced drags. The auxiliary thrust engines must overcome all of these drags. The low speed flaps down condition was also checked to assure that there was sufficient thrust for this condition. With the engine thrust levels required for the 300 knot condition, there is sufficient thrust for the low speed case.

The 300 knot test point requires a total thrust 9500 lbs for the RSRA aircraft. For two engines the thrust per engine is 4750 lbs. A thrust engine survey was conducted of production engines which would be capable of providing the propulsive force for the RSRA design condition. The results are shown below:

THRUST ENGINES	COUNTRY	STATUS	UNINSTALLED SEA LEVEL STD INTERMED. THRUST STATIC	SEA LEVEL STD INTERMED. THRUST 300 KNOTS
CF 700-2C	USA	Production	4120	3000
ALF 501	USA	Development	5100	3100
TF 34-GE-100	USA	Production	7915	5180
JT 15D	Canada	Production	2200	1370
J60-P-3	USA	Production	3000	2710
JT 8D-7	USA	Production	12600	10080
JD 8D-15	USA	Development	13700	10680
TFE 731	USA	Production	3500	2385
ATF 3	USA	Development	3750	2260
ASTAFAN 4	France	Development	2010	1100
Spey MK. 505-5	United Kingdom	Production	9750	NA
M45H	United Kingdom	Development	7760	NA

From the survey, the TF34-GE-100 engine was selected as the basic engine for the RSRA. A side mounted version of the TF34, the TF34-GE-100 is currently installed on the A-10A prototype. This mounting is the way it will be installed on the RSRA and it is assumed that the entire A-10A engine pod can be used for RSRA. If the A-10A does not reach production, an alternate version of the TF-34 could be used.

The General Electric TF34-GE-100 is a dual-rotor front fan turbofan engine with a bypass ratio of 6.23 to 1. The engine consists of a single-stage fan with a pressure ratio of 1.51 to 1, a 14-stage axial compressor with a pressure ratio of 14.5 to 1, an annular combustor, a 2-stage air-cooled axial gas generator turbine, and a 4-stage axial flow fan turbine driving the fan through a concentric shaft passing forward inside the gas generator rotor.

The sea level standard static ratings for the TF34-GE-100 engine without installation losses are as follows:

RATING	UNINSTALLED THRUST/TSFC	GAS GENERATOR SPEED	FAN SPEED	T_5 (°F)
Maximum	8985/.373	17600 RPM	7110 RPM	1495
Intermediate	7915/.362	17180 RPM	6720 RPM	1405
Max. Continuous	7260/.358	16910 RPM	6490 RPM	1350

The above ratings are based on the use of the -100 reference long fan exhaust duct and assume no accessory power loading or customer bleed extraction.

The TF34 design eliminates fan inlet guide vanes by introducing air directly to the fan rotor, a feature which contributes to guieter engine operation. A completely self-contained lubrication system is supplied with the engine. The high, 6.23 to 1, engine bypass ratio results in low thrust specific fuel consumption.

The TF34 control system consists of a fuel control, compressor variable-geometry control, temperature control, and automatic ignition control. The fuel control is basically a gas generator speed control scheduled by the power level except as modified by the temperature control to obtain the desired gas generator turbine discharge temp. The scheduling of speed and temperature is such that a low power speed is governing,and a high power temperature is governing. The TF34 design incorporates General Electric experience with the T64 series turboshaft engines and with turbofan designs over a wide range of thrust levels.

The TF34-GE-100 is utilized on the USAF/Fairchild A-10A prototype close support aircraft and has been selected for use on the USAF/Boeing AWACS aircraft. The similar TF34-GE-2, which differs from the -100 in being rated with a short fan exhaust duct and a different mounting structure,was chosen to power the U.S. Navy/Lockheed S-3A ASW aircraft. A photo of the Tf34-GE-2 is shown as Figure 27.

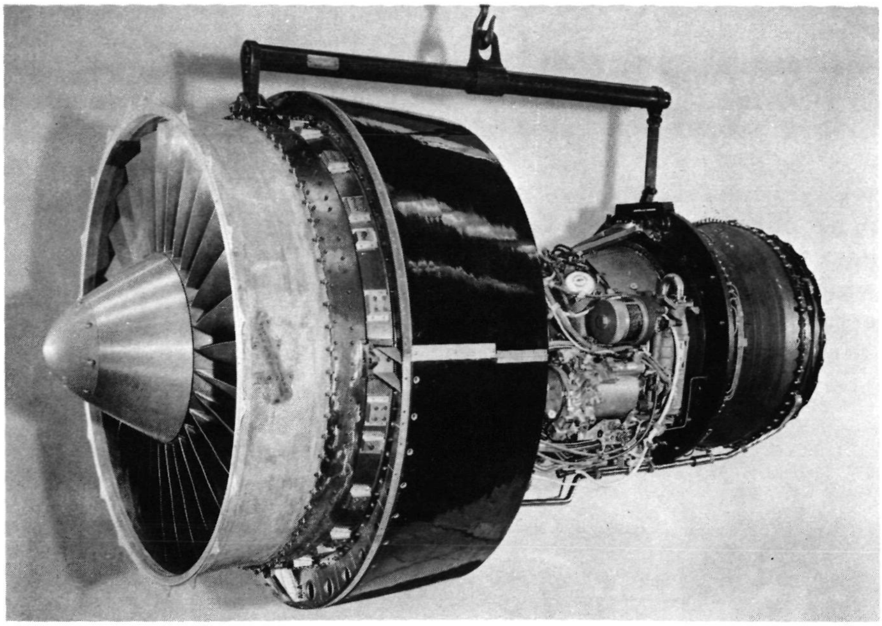


FIGURE 27 TF-34-GE-2 TURBOFAN

The TF34-GE-100 auxiliary propulsion engines are installed on the RSRA aircraft in pods identical to those utilized on the USAF/Fairchild Ind. A-10A aircraft. The use of pod-mounted auxiliary engines allows a minimum of localized pod-to-airframe interfaces which provide for ease of removal. The entire pod-mounted auxiliary engines are removed and replaced with a one piece fairing when auxiliary thrust is not required.

The inlet and exhaust losses of the TF34 are those associated with the A-10A pod design. The pod exhaust duct losses are already accounted for in the -100 ratings since the latter include the long reference fan exhaust duct installation. The pod installation losses are estimated at 2 percent thrust.

The TF34 engines and pods are mounted to the aircraft via front and rear mounts which react to vertical, horizontal, and torque loadings. Fore-and-aft restraint is provided by a thrust reactive member incorporating a load cell for measuring auxiliary propulsion engine thrust.

The auxiliary propulsion pods basically consist of TF34-100 engines, mounts, cowling, exhaust system, and lube system. In addition, parts of engine controls; and starting, fuel, and fire extinguishing systems are included. The total weight of both pods is estimated at 3823 lb. Allowing 3 lbs. for capping lines yields a net weight change of 3820 lb. when these pods are removed for hover testing.

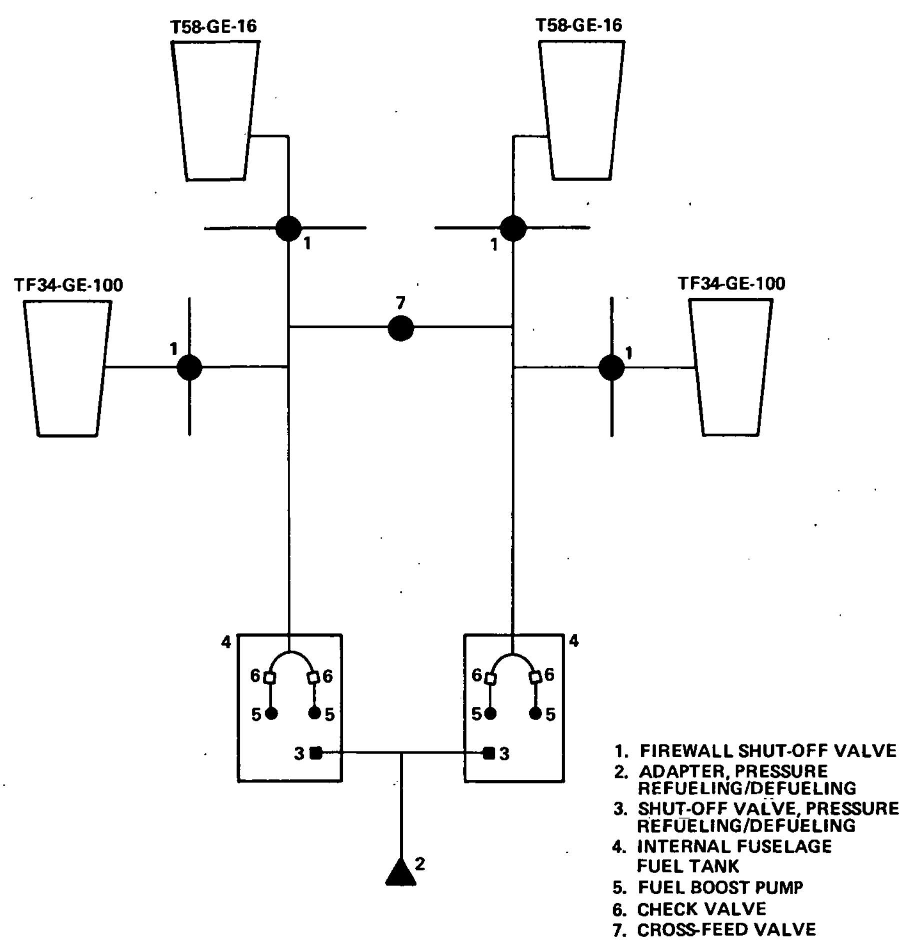
Starting with the compound configuration at a design gross weight of 26392 lb, the removal of the two auxiliary propulsion pods at station 277 will decrease the gross weight by 3820 lb. and shift the horizontal center of gravity 3.0 in. aft.

Fuel System

The RSRA fuel system (Figure 28) supplies pressurized fuel from two fuselage tanks to the General Electric T58-GE-16 engines and to the General Electric TF34-GE-100 auxiliary propulsion engines when they are installed.

The fuel supply consists of two internal fuselage tanks with a combined capacity of 769 gallons of JP-4 fuel. Two separate feed systems are provided such that one tank feeds the right T58 and TF34 engines, while the other feeds the left hand engines. A cross-feed capability is incorporated so that all four engines may be fed from either tank providing for the use of all fuel onboard. Pressurized fuel feed is provided by two independent fuel boost pumps located in each tank.

Pressure refueling/defueling and gravity refueling capabilities and provided.



RSRA FUEL SYSTEM

FIGURE 28

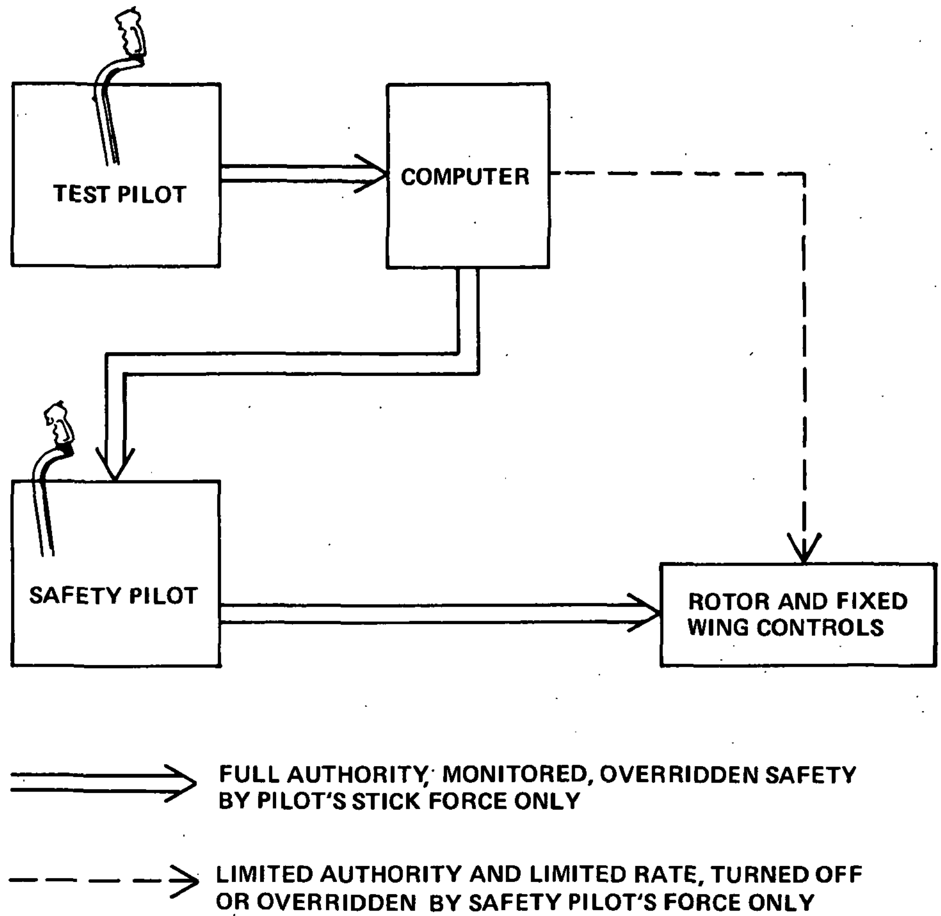
Flight Control System

The RSRA Flight Control System has been designed to provide the basic characteristics required of a rotor research aircraft on a system requiring minimum development. The basic system characteristics include:

- Separate fixed wing and rotary wing controls
- Control of an in-flight variable incidence wing
- Control of wing flaps and leading edge lift devices
- Control of an in-flight variable drag device
- Control of tail fan with shutters for high speed flight
- Capability for autopilot operation, preprogrammed testing and helicopter simulation

The control system design concept selected achieves the RSRA objectives by separating the functions of the two-man crew. The pilot (in the right hand station) performs the function of test pilot. His primary function is conducting the particular flight test of interest at the time; high speed testing, performance mapping, simulation, etc. The co-pilot at the left hand station is a safety pilot. His primary function is to monitor the aircraft status for safe operation. He can return the aircraft to normal operation and land from any anticipated test condition.

In order to provide the flexibility for the test pilot and monitoring capability for the safety pilot, a control system was devised which channels the test pilot's commands through a computer to the safety pilot's control sticks and then to the primary flight controls. Primary control of the aircraft is then monitored by the safety pilot as his sticks are moved by the test pilot. Exceptions to this concept are limited authority and limited rate movements of some of the controls by the computer in response to preprogrammed relations. In all cases where the exception occurs, the resulting control system movements can be turned off and/or overridden by the safety pilot. A block diagram of the control system concept is shown as Figure 29.



SCHEMATIC OF RSRA CONTROL SYSTEM CONCEPT

FIGURE 29

The requirement that the aircraft simulate a wide variety of rotor/vehicle configurations with a high degree of dynamic fidelity places considerable emphasis on the flight control system design. To simulate, with one vehicle, various aircraft with different aerodynamic performance characteristics requires that the rotor be controlled separately from the fixed wing surfaces. The control integration scheme needed to integrate the various control elements must include variable system gains and essentially different control signals for each control surface or element. A mechanical control system capable of providing the proper control functions and the flexibility would be costly, heavy, and complex. A completely fly-by-wire system would be the optimal solution, but there is a lack of currently qualified and available hardware and the cost of such a system is much more than a mechanical system. During the RSRA Predesign Study, an all fly-by-wire system and an electrical/mechanical system were designed and compared. Both systems perform the aforementioned tasks and are discussed later.

The RSRA flight control systems contain several features which are included to provide either required or necessary functions of the RSRA. Listed below are the major features designed in the control system and the functions they perform.

FAS (Force Augmentation System)

This system consists of sensors, a computer and actuators in series with the test and safety pilot's sticks which provide:

1. Fixed wing type stick force cues, making maneuvering flight easier and more precise.

2. The route for the input from the test pilot to the safety pilot's stick for safety pilot monitoring.

SAS (Stability Augmentation System)

A limited authority system which improves flying qualities by attitude stabilization via a gyro, amplifier, shaping and servo system. On the RSRA, this system will be used primarily at low flight speeds by the safety pilot with the computer off.

COMPUTER

The RSRA computer will be capable of transmitting the pilot's commands to the safety pilot's controls, and with limited authority and limited rate to the primary flight controls. It is capable of providing autopilot operation, preprogrammed testing and helicopter simulation. Accurate measurements of rotor forces and moments and wing forces will be available to the computer. The capabilities of the aircraft in the above modes is discussed in the section on helicopter simulation and model following, page 144.

ROTOR-FIXED WING INTEGRATION UNITS

The sensitivity of the rotor and fixed wing control surfaces relative to the pilot inputs and each other should be variable for basic research and for high speed flight where the rotor and fixed wing controls are very sensitive to stick inputs. Rotor/fixed wing control integration units are provided to perform this function. These devices can be positioned to lower rotor sensitivity at high speeds without lowering the fixed wing control sensitivity. The mechanical linkage which provides this task is shown on the flight control schematic, Figure 30.

CONFIGURATION CONTROL PANEL

The RSRA has several unique inflight positioning features to allow a particular test configuration. These features are set by the configuration control panel. The panel is a bank of levers in the cockpit which provide the pilots with direct control over the following devices.

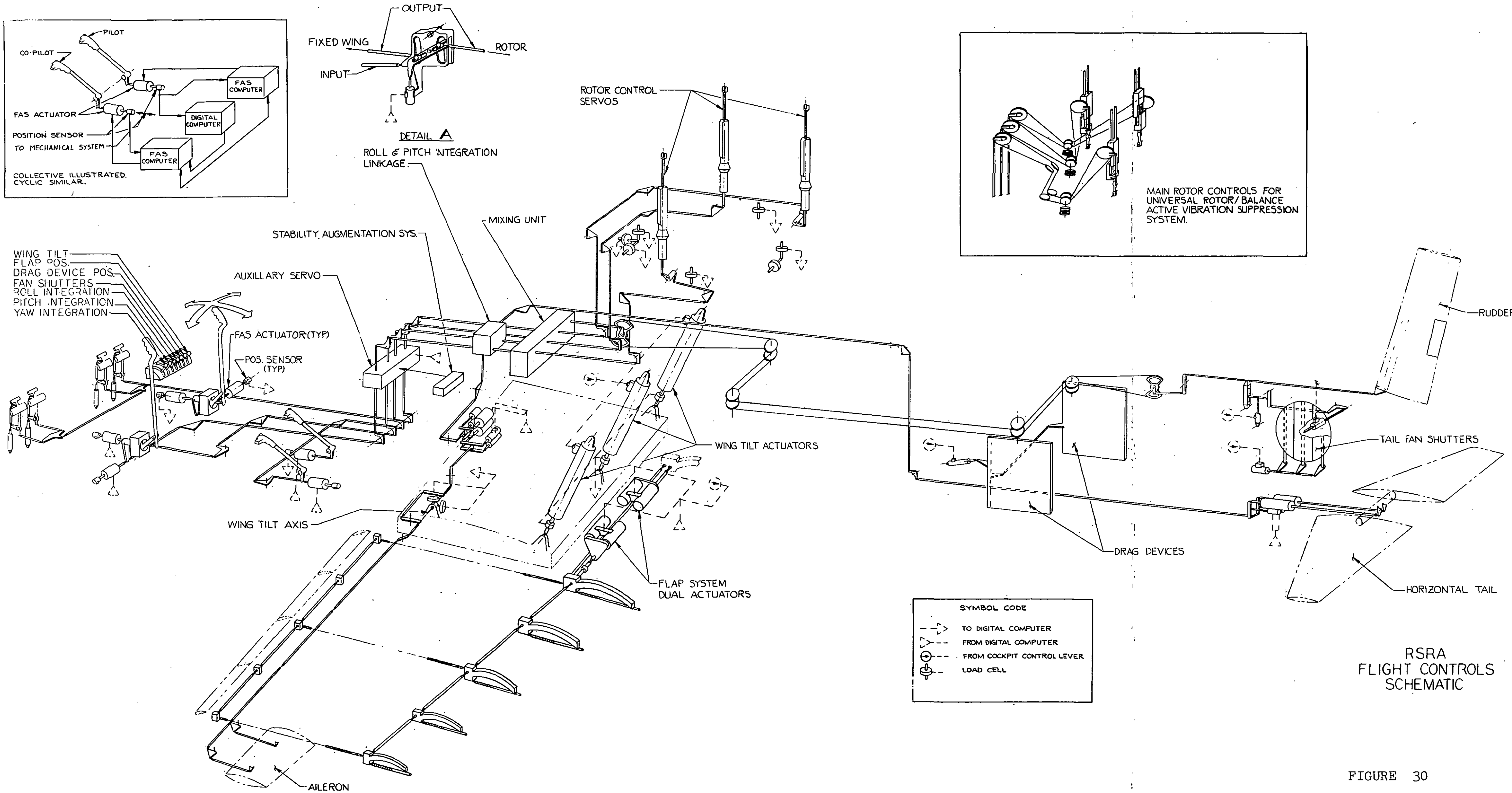
- Pitch, yaw, and roll control sensitivity devices
- Tail-fan shutter position
- Drag device position
- Flap position
- Wing incidence angle

Electrical/Mechanical Flight Control System

Description of Mechanical Portion of the System

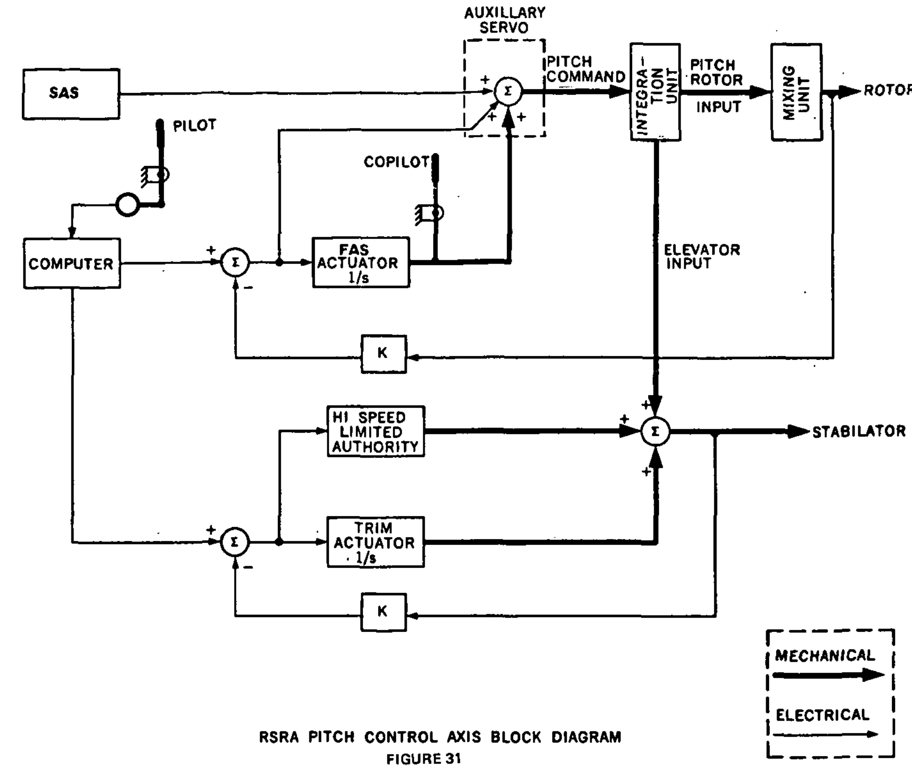
COPILOTS CONTROLS -

The copilot has conventional mechanical controls as shown in Figure 30. The cockpit arrangement consists of a center mounted cyclic control, a collective control mounted to the left of the seat, and floor mounted rudder pedals. A four channel, S-61 auxiliary servo provides power boost for rotor control inputs received from the cockpit controls and is capable of reacting flight control loads. The auxiliary servo also provides a limited authority electrical input path to the flight controls. The integration units, detail A in Figure 30, receive inputs from the auxiliary servo. These units apportion the control inputs between the rotor and the fixed wing control surfaces and can be set by the pilot or the computer to vary the rotor/fixed wing control deflection ratio. This unit is ground adjustable to allow variation in the ratio gradients and limits as required for various rotor types and configurations. An S-61 mixing unit converts the rotor control inputs from the integration units to rotor control inputs to the primary servos. It also provides the required collective to yaw coupling to minimize heading change during power adjustments. Three S-61 primary servos receive control inputs from the mixing unit to position the swashplate and effect control. The fixed wing outputs from the integration units are mechanically connected to the control surfaces through the surface actuators. Each surface actuator consists of a primary boost servo, series trim servo, and a high speed series servo. It is dual and monitored since it has full authority, series capability which cannot be manually overridden by the pilot. The electronics controlling the trim servo are also dual as are the hydraulic supplies. The high speed servo is limited in authority to prevent sudden large inputs from endangering the aircraft. This control configuration is shown in the pitch axis block diagram, Figure 31, which is typical of the control axes.

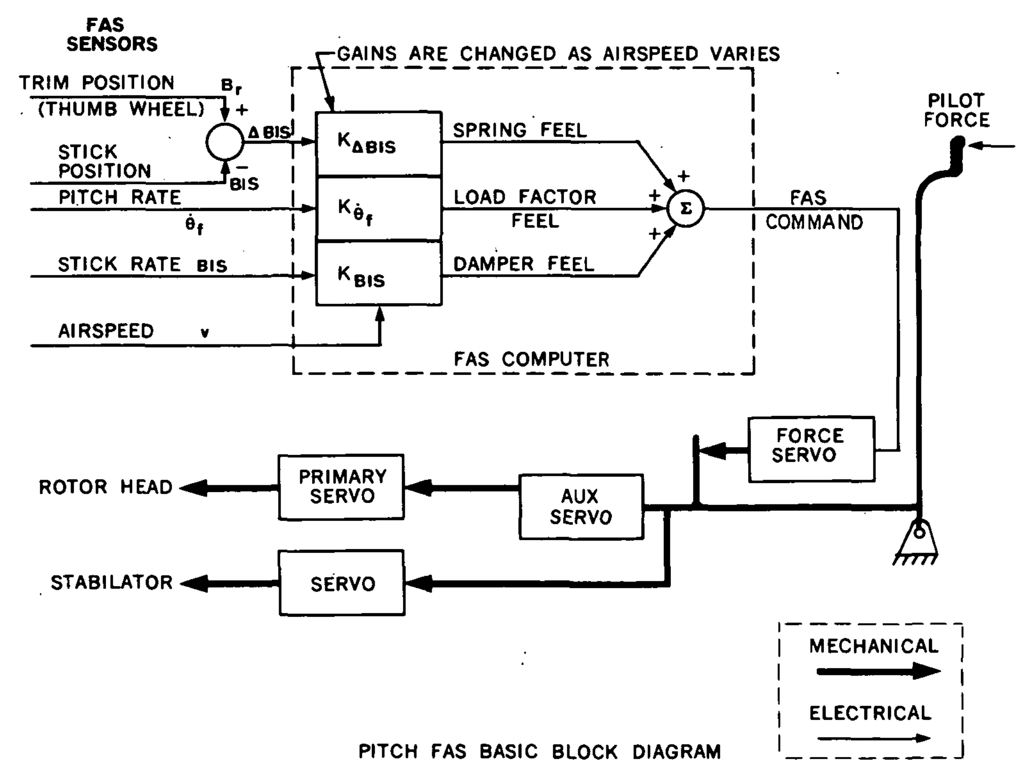


RSRA
FLIGHT CONTROLS
SCHEMATIC

FIGURE 30



RSRA PITCH CONTROL AXIS BLOCK DIAGRAM
FIGURE 31



PITCH FAS BASIC BLOCK DIAGRAM
FIGURE 32

STABILITY AUGMENTATION SYSTEM (SAS)

A three axis SAS is used to provide additional damping and handling qualities improvement during hover and low speed flight. Pitch, roll, and yaw rate gyros sense the aircraft rates which are then shaped and fed back to the flight controls through the limited authority auxiliary servo, as shown in Figure 31. These signals are faded out and removed as the aircraft speed increases beyond 80 knots and the aerodynamic surfaces become more effective. The SAS is intended to assist the copilot in controlling the aircraft. During computer controlled testing, the SAS is temporarily turned off and will come on line whenever the copilot manually overrides the computer control.

FORCE AUGMENTATION SYSTEM (FAS)

Maneuvering control force cues required for high speed flight are provided by the FAS. A three axis system, pitch, roll and collective is provided for both pilots. The pitch FAS provides the pilot with longitudinal cyclic control forces which are proportional to load factor, stick rate, and stick deflection. Lateral cyclic control forces are proportional to stick deflection and stick rate and are scaled to provide control harmony. The collective FAS provides a force proportional to rotor control loads and a collective stick vibration as the loads reach a predetermined level. These force cues will provide the pilot with the cues necessary for maneuvering safely throughout the useable flight envelope. The FAS actuators are high speed, full authority actuators to allow the pilot to make rapid stick motions without feeling variations in the applied stick force. As a result, each of the components of the FAS are dual and the output of the dual actuators monitored to provide fail safety. The force actuators each apply half of the force to the control through a pivot bar linkage. If one of the force paths should malfunction, the force produced by the actuators will differ and the pivot bar will move from the vertical. This movement is sensed and the system is shut down.

Description of Electrical Portion of System

PILOT'S CONTROLS

The pilot's cockpit controls are identical in form and location to the copilot's controls. His collective and cyclic controls, however, are mechanically disconnected from the copilot's controls and the aircraft flight controls. His control inputs are sensed electrically and sent to the copilot's FAS where a command is initiated to change the trim reference of the copilot's control. The motion of the copilot's control to its new trim reference produces the necessary mechanical control response. Similarly, the copilot's control motion is sensed and sent to the pilot's FAS to allow his controls to track the copilot's when the copilot has command of the aircraft.

COMPUTER CONTROL

The computer interface to the aircraft flight controls is through the copilot's mechanical system. Figure 31 shows the computer interface. Computer inputs to the rotor are compared to the present rotor control position and the error is sent to the copilot FAS and the limited authority auxiliary servo. The FAS is configured as an integrator in the computer control mode and, as such, will drive the error to zero. The high speed auxiliary servo is a position servo which will reduce the error immediately while the FAS is eliminating the error at a limited rate. The final result is a system which will provide faithful reproduction of the computer commands at the rotor. The computer control of the fixed wing surfaces is identical to that of the rotor with the dual trim actuator performing the function of the FAS actuator.

CONFIGURATION CONTROL PANEL

The position of the RSRA configuration control, consisting of the drag brakes, and wing incidence control levers, are controlled electrically from the levers mounted on the Configuration Control Panel. Also, controlled from this panel are; flap trim, rudder sensitivity, pitch and roll sensitivity and tail fan shutter position. The controls of this panel are available to both pilot and copilot.

Wing Control System Devices

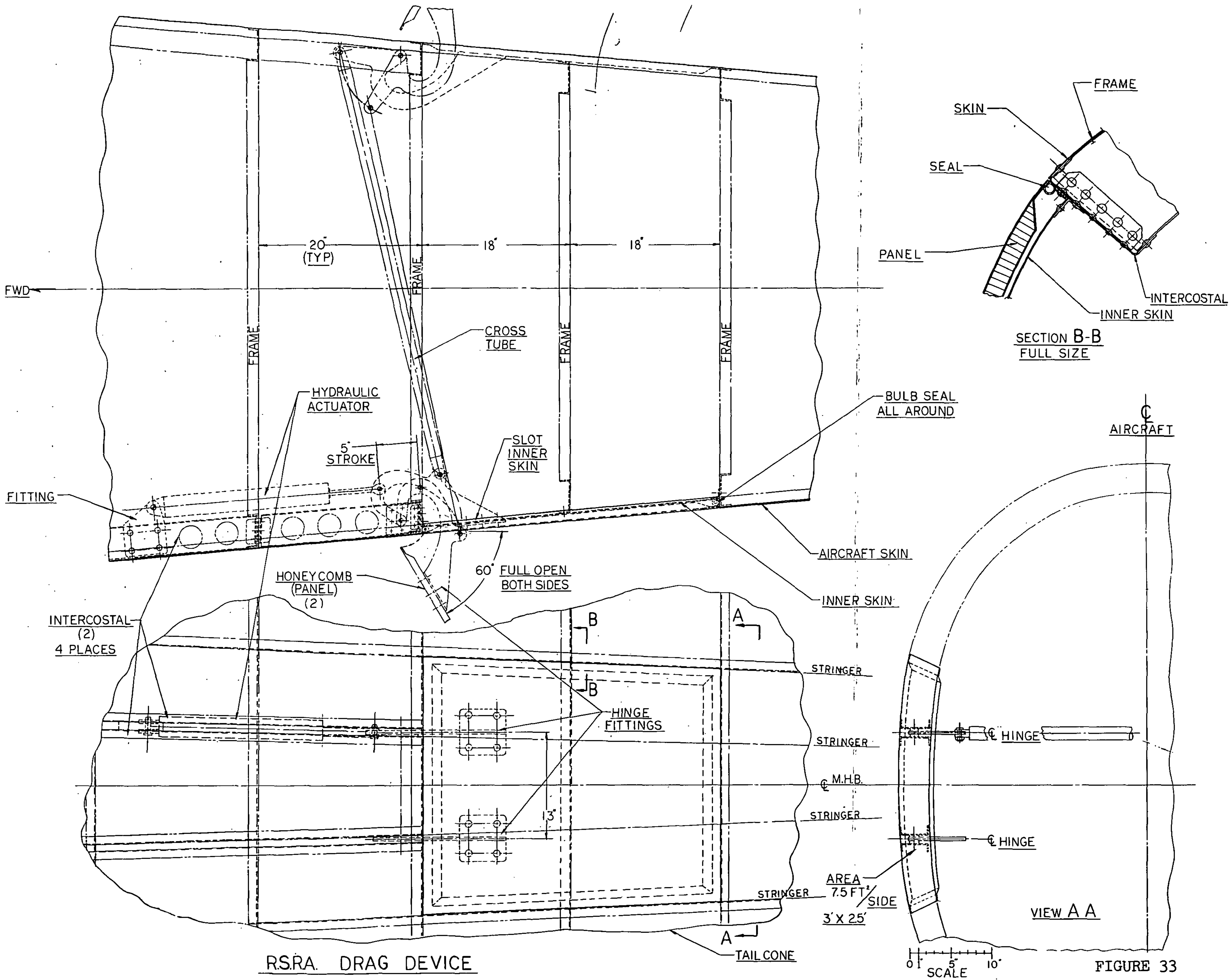
The major control devices used on the wing are the high lift devices, the wing variable incidence mechanism and the ailerons.

The high lift devices on the large wing consist of leading edge slats and trailing edge double slotted flaps. The leading edge slats are extended only at high flap deflections in the landing configuration and at simulation conditions that require almost complete rotor unloading at low speeds near 100 knots. The trailing edge flaps are used as primary lift control for simulation and are actuated by high speed limited authority actuators for this condition. The range of the trailing edge flaps is zero to 60 degrees and the leading edge slats range from zero to 25 degrees.

Both wing configurations of the aircraft have inflight variable incidence. The incidence is varied by three hydraulic actuators which are controlled by a lever in the cockpit. The actuators are designed to provide the full incidence range required by the wing in order to achieve ±10 degrees of effective rotor shaft tilt by varying fuselage incidence. This actuator range is 42 degrees. The drawing of the wing tilt actuators is shown as Figure 14, page 35.

Drag Brakes

The split plate drag brake, located on the sides of the aft fuselage was selected because in this position there was enough brake area available, and the design yielded good test flexibility, a minimum of undesirable moments and



R.S.R.A. DRAG DEVICE

FIGURE 33

comparatively easy structural integration. The installation of the brake and its actuation is shown as Figure 33 . The total deflection is 60°.

The brakes on each side of the aircraft are extended by a single actuator. The brake position is set by the pilot by a lever on the configuration control panel.

Optional Electrical Control System Description

The electrical, or fly-by-wire, control system option for the RSRA uses the flexibility of the fly-by-wire concept to simplify the control layout. A total of 6 actuators, two integration units, a FAS system, major portion of the mixing unit, and the interconnecting mechanical links of the baseline system are replaced by 5 actuators and an electronic control unit. Figure 34 shows the block diagram of the electrical control system. The cockpit controls are identical to those in the electrical/mechanical system. The pilot and copilot controls are connected mechanically and they share a single FAS. The FAS in this system is used only to provide maneuvering control feel since the electronic control actuators provide full authority control inputs at the control surfaces. The control motions of the cockpit are sensed and sent to electronic control units. These units perform the control integration and mixing and provide the necessary electrical power to drive the actuators. All of the electronic paths are quadruply redundant. Each channel is monitored and majority is used to provide the system with mission reliability equal to or greater than that of a conventional control system.

The electronic control actuators are the interface between the system electronics and the control surfaces. These actuators are positioned by electrical command signals from the electronic control unit. They are not capable of reacting the flight loads of the RSRA but supply mechanical inputs to the conventional main rotor primary servos and fixed wing control surface actuators. The full authority high speed capability of these actuators requires that they be redundant and that their performance be monitored to prevent malfunctions from endangering the aircraft. The actuators are quadruply redundant and provide the ability to sustain two failures without loss of system performance.

Computer Control

A major item in the RSRA concept is the use of the onboard digital computer for control and simulation. The digital computer interfaces with the RSRA control system through the FAS, the auxiliary servo, and the control surface actuators as shown in the pitch block diagram of Figure 31 . The pilot and copilot have data entry and mode control panels available to them from which they can control the computer operation.

FLY-BY-WIRE CONTROL SYSTEM

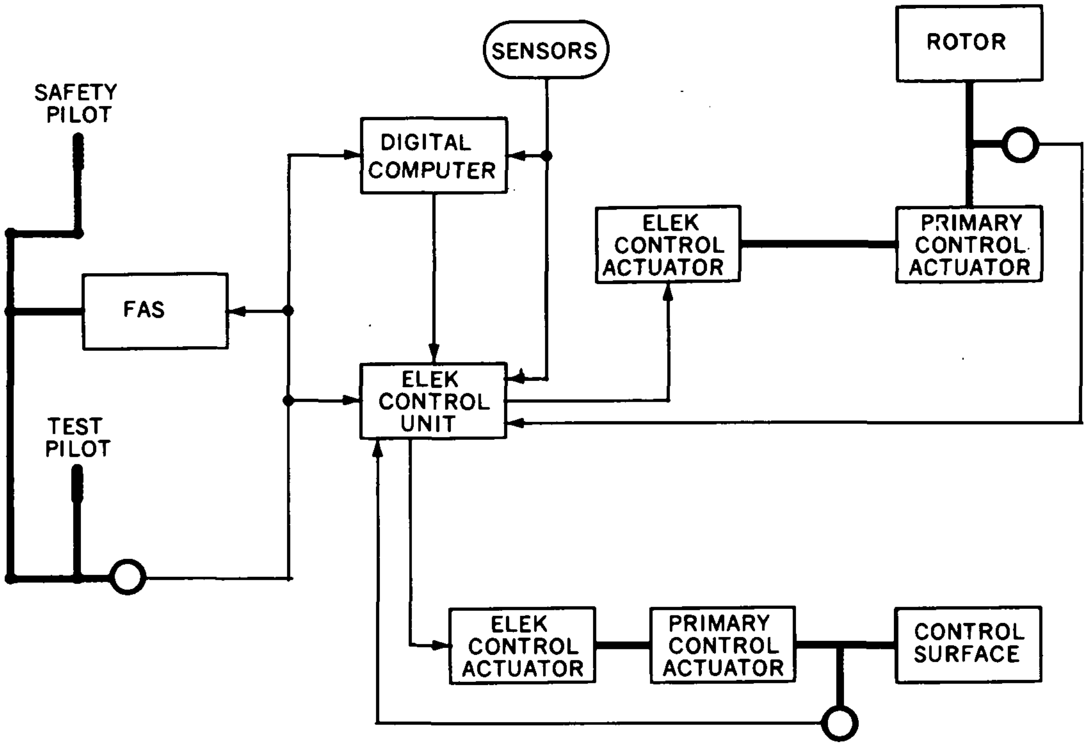


FIGURE 34

AUTOPILOT OPERATION

The computer may be programmed to serve as an autopilot for the RSRA. The required sensors interface directly with the computer. The various autopilot modes are selected and controlled from the crew's computer control panels. The computer interface with the flight controls is through the copilot's FAS. Computer commands are reflected as motions of the crew's control sticks which allow monitoring of computer performance.

INDEPENDENT ROTOR-FUSELAGE CONTROL

Helicopter simulation, rotor testing and model following will require independent control of the rotor and the fuselage. This is provided in the RSRA through the digital computer. The rotor parameter measurements system will supply the rotor forces, moments and angles required for rotor control. The load cell measurements are processed in the digital computer and the resultant rotor parameters are used in the control feedback mechanization. The values of the rotor parameters are compared to the desired or commanded values and the error is determined. The error is then shaped and sent out of the computer and fed back to the controls. The signal paths are shown in Figure 31. Rotor commands are sent to the copilot FAS and the auxiliary servo.

The command signal is first compared to the rotor control position and an error signal is formed. The error is sent to the auxiliary servo to reduce the error in a limited authority, high speed manner. The remaining error is integrated in the FAS until the rotor control position is equal to the command and the error is zero. This configuration provides high fidelity full authority control with limited authority and limited rate actuators by effectively utilizing the virtues of each. The fixed wing control surface position is controlled in the same manner. In this configuration, the mechanical input from the integration unit is compensated for in the error control loop.

SINGLE PILOT OPERATION

The RSRA may be flown throughout the flight envelope by either the pilot or the copilot. Both pilots have access to complete set of flight and auxiliary controls. A failure of the copilot's FAS requires the copilot to resume direct mechanical control of the aircraft.

Auxiliary Systems

RSRA Electric Power System

The RSRA has a prime dc generating system with ac power derived from static inverters. A block schematic diagram is shown as Figure 35.

Two self-cooled, 300-ampere dc generators are mounted on and driven by the main rotor transmission. These generators have been in use on Sikorsky S-61 helicopters for several years. They can provide the full rated load of 300 amperes at 28 V dc over the speed range of 4000 to 8000 rpm. 200 amperes are available at approximately 24 volts for speed down to 3200 rpm - equivalent to 40% rotor rpm. The generators operate in a simple, split-bus distribution system in which the sources are never parallelled. Reverse-current cutouts are therefore unnecessary.

A dc essential bus is provided that can be powered from any one of three sources - the dc primary busses or the battery. The 22 ampere-hour nickel-cadmium battery is protected from overcharge and overtemperature by means of a battery charger. In the event of a malfunction of one generator, the remaining unit will power both primary busses. Total loss of generator power will result in the essential bus being powered by the battery.

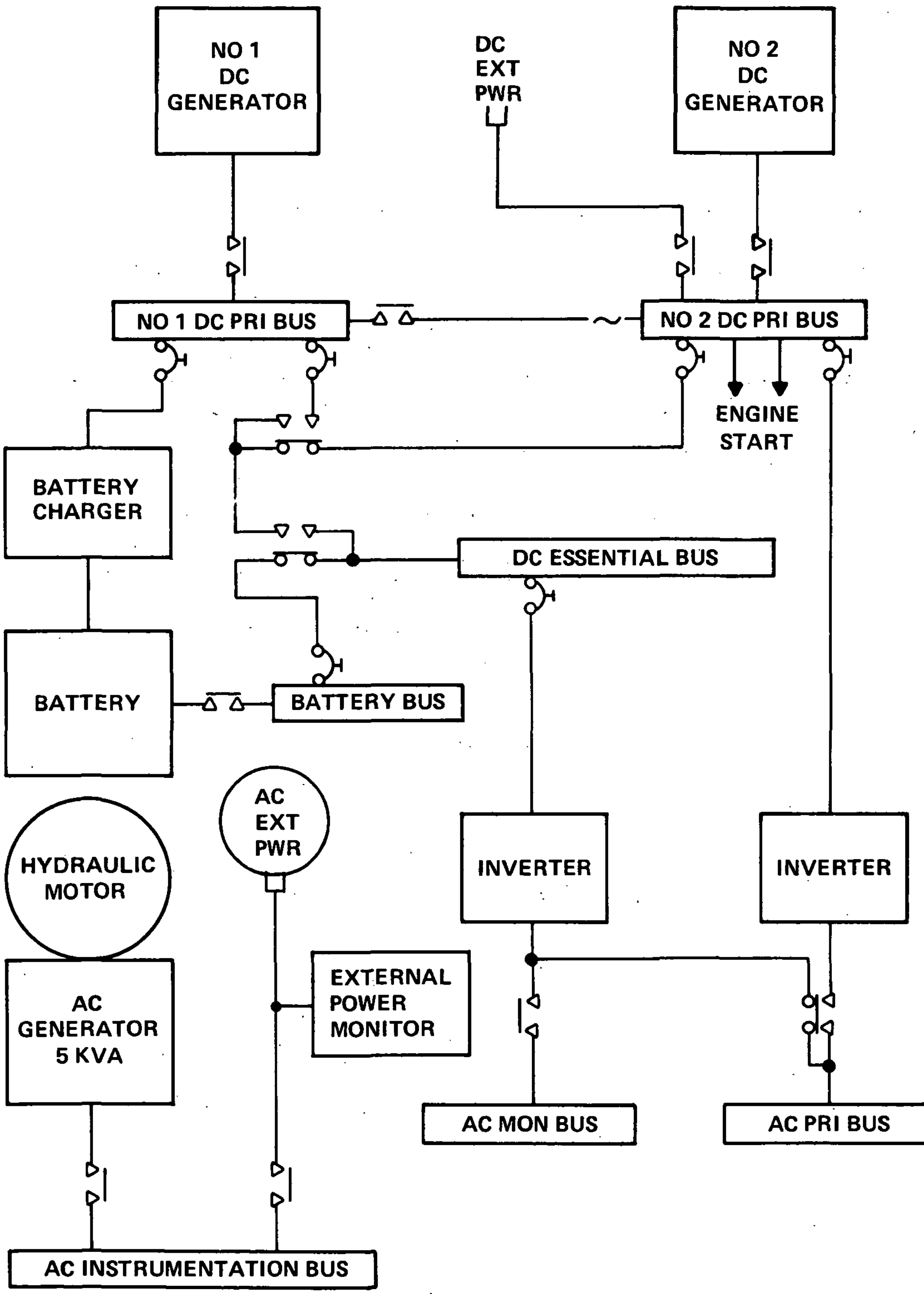
AC supplies for normal aircraft loads are provided by two 750 VA static inverters. Loads essential to flight are connected to the ac primary bus. Non-essential loads on the ac monitor bus are dropped in the event of a malfunction causing loss of one inverter output.

The T58-GE-16 engines are started electrically using power drawn from the dc primary bus system. They are started by dc external power, or from the dc generators for airborne restarts.

AC power for the research instrumentation and digital computer is supplied by a 5 KVA, 3-phase, 400 Hz generator driven at constant speed by a hydraulic motor. This unit is in production for the F-14A aircraft. The instrumentation bus can also be powered from ac external power for ground check-out or calibration. An external power monitor unit protects the aircraft equipment from reverse phase-sequence, over-voltage, under-voltage, and under-frequency faults on ground power.

Hydraulic System

The design goal of the hydraulic system is to provide the maximum amount of aircraft safety, flexibility, system reliability and redundancy while meeting the system requirements with the minimum number of separate systems. In order to provide these attributes the hydraulic power system is divided into



RSRA - ELECTRIC POWER SYSTEM

FIGURE 35

two major systems. The first major system provides power to the helicopter flight control system. It has two subsystems (Systems 1 and 2) which have completely independent hydraulic power sources. The second major system supplies the fixed wing flight controls and aircraft utility systems. This system also is composed of two completely independent hydraulic power sources (Systems 3 and 4). A fifth minor system is included as a separate and integral unit of the rotor brake. It has an electric motor driven pump system. This arrangement meets the requirements and provides separation of the two major control systems for flexibility. Furthermore, it allows for the shortest line routing for each system, thus decreasing the number of potential leakage points.

The configuration results in primary functions being powered by redundant actuators, each supplied by a separate hydraulic power source. Secondary actuators and utility functions are supplied from one of the fixed wing power systems through a priority valve. This valve insures that an adequate power level is available first to the primary control actuators whenever both types of functions occur simultaneously.

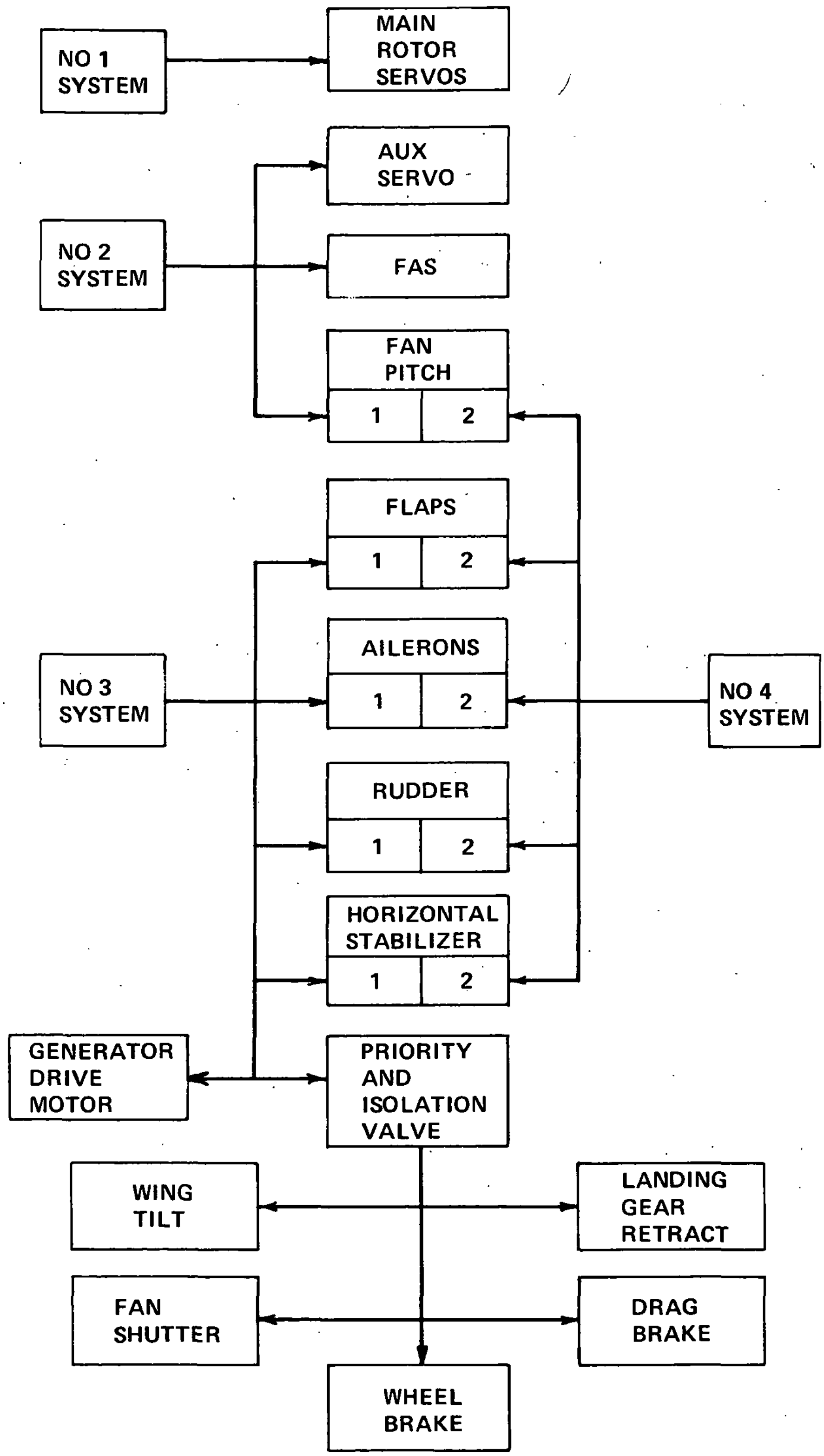
Power for the four subsystems is derived from gear-driven, variable delivery, pressure compensated pumps. All gear driven pumps are powered whenever the main rotor head is turning. A reservoir is provided for each hydraulic system. The reservoirs are of the vented type, rated for Class I type operation as per MIL-H-5440. MIL-H-5606 hydraulic fluid is used throughout. Filtration is provided in the pressure and return lines, and is in accordance with MIL-F-8815. External ground-test connections are provided on all systems, except the rotor brake-system. The rotor brake system is a self-contained electric motor driven integrated module. Pressure gages and word/caution lights are provided for each hydraulic system. Remote reservoir level readouts are provided on all four systems which give cockpit indication of system fluid levels. Figure 36 is a block diagram of the power system arrangement.

Number 1 System

The Number 1 system is a 1500 psi class Type I system as defined by MIL-H-5440. Power is supplied by a variable delivery piston type pressure compensated pump mounted on the accessory drive section of the main rotor transmission. Drive power is maintained so long as the rotor is turning, even when the engines are not operating.

The primary servos react the main rotor aerodynamic loads. The auxiliary servo provides a redundant capability as a back up for the primary servos.

A pressure sensitive electrical interlock between the primary and auxiliary servos prevents turn off of either system if the pressure in the other system is low. The turn off valves are normally open solenoid valves which require an electrical signal to close. Thus, hydraulic power to the servos will not be lost in the event of an interruption in electrical power.



BLOCK DIAGRAM OF SYSTEM DISTRIBUTION

FIGURE 36

Number 2 System

The Number 2 system is a 1500 psi class Type I system as defined by MIL-H-5440. The pump and power supply drive is the same as that of the primary servo system. This system services the auxiliary servo, the Force Augmentation Servo (FAS) and one stage of the tail fan pitch control servo. In addition to the interlock circuit with the main rotor primary servo system, a similar interlock is provided with the portion of the Number 4 system which supplies power to the second stage of the fan pitch actuator.

Number 3 System

Power for the Number 3 system is generated by a 3000 psi class piston pump. The drive arrangement is similar to that of the primary servo system for ease of ground maintenance and in-flight integrity of the flight controls. The No. 3 system supplies power to the first stage of the wing-flap, aileron, rudder and the horizontal stabilizer servos. In addition the No. 3 system powers the utility functions; the drag-brake, the landing gear retract mechanism, the fan-shutter, wing-tilt, the wheel brakes, and the constant speed generator drive motor. The utility functions are connected to the No. 3 system through a priority and an isolation valve, thus assuring that sufficient power is available first to the flight control system. The priority valve utilizes the pressure flow characteristics of the system pump to regulate flow to the utility group. If the system pressure drops, the priority valve will limit flow to the utility function. The isolation valve is normally closed in flight and opens only when a specific utility function is commanded. A scissor switch on the landing gear provides a signal to open the valve whenever the aircraft is on the ground.

Velocity fuses and return line check valves are installed in each branch of the utility system, thereby preventing rapid reservoir depletion in case of a utility failure.

Number 4 System

Power generation is the same as that of Number 3 system. This system is also a 3000 psi class Type I as defined by MIL-H-5440. The No. 4 system supplies power only to the second stage of the wing-flap, rudder, aileron, horizontal stabilizer, and tail-fan servos.

System Power Requirement

The hydraulic system power requirements were established by estimating the system loads and control surface rates. Table IX shows the system rates and the corresponding flow demands.

The total system power requirement is predicated on simultaneous inputs into all flight control channels, on constant flow demand of the generator drive motor, and on a single utility function. It is assumed that operation of the wing-tilt, drag-brakes, landing-gear, and fan-shutters does not take place concurrently. Table X gives the continuous system flow demand. Table XI shows the maximum total requirements per system and the power available.

TABLE IX

SYSTEM FLOW DEMANDS

Component	Rate Requirement	Flow Demand (GPM)
Primary Servo	100%/sec.	4.5
Aux. Servo	100%/sec.	4.5
Flap Servo	100%/sec.	7.25
Aileron Servo	30°/sec.	.81
Rudder Servo	30°/sec.	.364
Horiz. Stab. Servo	30°/sec.	.445
Tail-Fan Servo	100%/sec.	1.15
Tail-Fan Shutter Servo	.1 in/sec	.025
FAS Servo	1.43 in/sec	1.08
Wing Tilt Actuators	1.4°/sec	5.43
Landing Gear Retract Mechanism	.86 in/sec	1.5
Drag Brakes	6°/sec	.485
Gen. Drive Motor	Continuous	5.75

TABLE X
CONTINUOUS SYSTEM FLOW DEMAND

Component	System Demand (GPM)			
	No. 1	No. 2	No. 3	No. 4
Primary Servos	4.5			
Auxiliary Servos		4.5		
FAS		1.1		
Flap			7.25	7.25
Aileron			.8	.8
Rudder			.4	.4
Horiz. Stabilizer			.45	.45
Tail-Fan Pitch		1.15		1.15
Wing Tilt			5.50	
Generator Drive			5.75	
Total	4.5	6.75	20.15	10.05

TABLE XI
MAXIMUM TOTAL SYSTEM REQUIREMENTS

System	Flow Requirement (GPM)	Flow Available (GPM)
1	4.5	6.5
2	6.75	7.5
3	20.15	22.00
4	10.05	12.00

Avionics

The RSRA avionics have been selected to meet the government's technical requirements. Readily available existing equipment, in government inventory, has been chosen for all systems. The location of the avionics is in the nose compartment of the aircraft; their controls are on the cockpit center console.

Communications

Communications equipment consists of one AN/ARC-115 VHF/AM radio set, one AN/ARC-116 UHF/AM radio set, and one C-6533/AIC ICS. This ICS is provided for three stations, the pilot, the copilot, and the instrumentation engineer.

Navigation/Identification

Navigation/Identification equipment consists of one AN/ASN-43 gyro-compass, one AN/ARN-82AVOR/LOC navigation system, one R-844/ARN-58 glide slope/marker beacon, one AN/ARN-52 TACAN, plus an AN/APX-72 IFF transponder.

Telemetry Data Link

In addition to the equipment specified above, various avionics involved with the telemetry data link are included. These are discussed in the section on test instrumentation.

Cockpit Environment

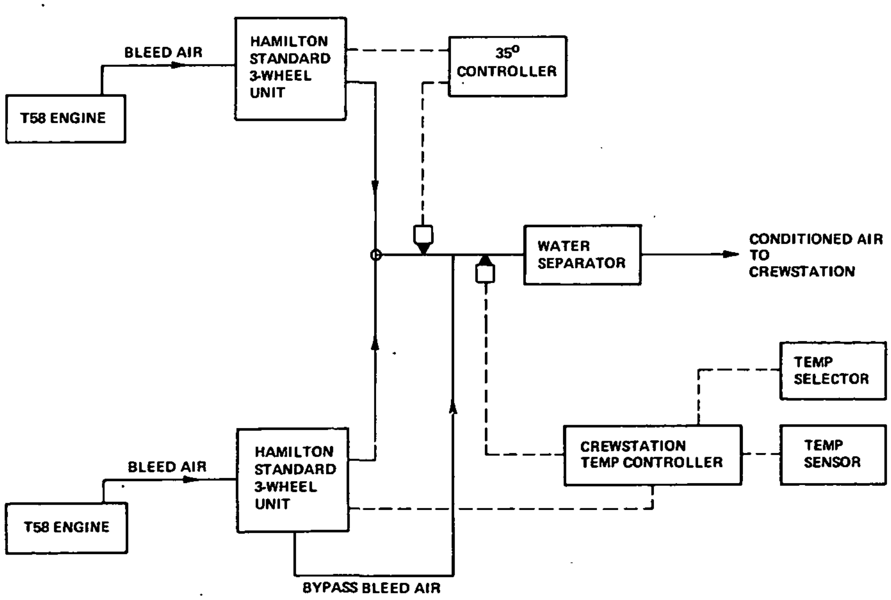
The environmental control system (ECS) for the RSRA aircraft is designed to provide a crew station air temperature of 85°F at sea level, 110°F ambient conditions. In the heating mode, the system will provide a crew station air temperature of 65°F with a 0°F ambient temperature to an altitude of 15000 feet. Crew station ventilation provisions are incorporated for use in the event of an environmental control system malfunction.

The RSRA ECS system consists of two Hamilton Standard 3-wheel bootstrap air conditioning units as employed on the Sikorsky S-67 Blackhawk helicopter. A single unit is used on the Sikorsky S58T Helicopter.

The system (Figure 37) operates on bleed air extracted from both General Electric T58-GE-16 engines. The bleed air enters each 3-wheel unit and is partially cooled through a primary air-to-air heat exchanger. The bleed air then enters a turbine driven compressor where its pressure and temperature are boosted from which it then passes through a secondary air-to-air heat exchanger decreasing its temperature. Additional cooling is achieved by expansion through the compressor turbine where pneumatic energy is converted to mechanical energy. The turbine also drives an air-fan which provides the cooling airflow through the heat exchangers. The temperature of the refrigerated air leaving the turbine is regulated to the desired level by the mixing with engine bleed air bypassing the refrigeration cycle. The quantity of hot air bypassed is regulated by the crew station temperature controller. The resulting mixture passes through a water separator where moisture is removed and then on to the crew station. The moisture collected in the water separator is sprayed into the heat exchanger cooling air inlet providing further cooling.

In the RSRA application, one of the two units operates at maximum refrigeration while the second unit's output is regulated to produce a temperature of 35°F when the two outputs are mixed. The crew station temperature controller regulates the mixture temperature above 35°F by utilizing bypass bleed air from the first unit. The resulting mixture at the required temperature then passes through the water separator and into the crew station. The ECS bleed air requirements are within the bleed flow capability of 3 percent of engine airflow of the T58-GE-16 engine under RSRA operating conditions.

Crew station ventilation is provided in the event of an ESC malfunction by a manually operated ram air intake and an auxiliary blower. Air enters the ram air intake through a spring loaded flapper valve, circulates through the crew station, and is expelled through the blower. A check valve at the blower exit prevents reverse flow.



ECS SYSTEM SCHEMATIC

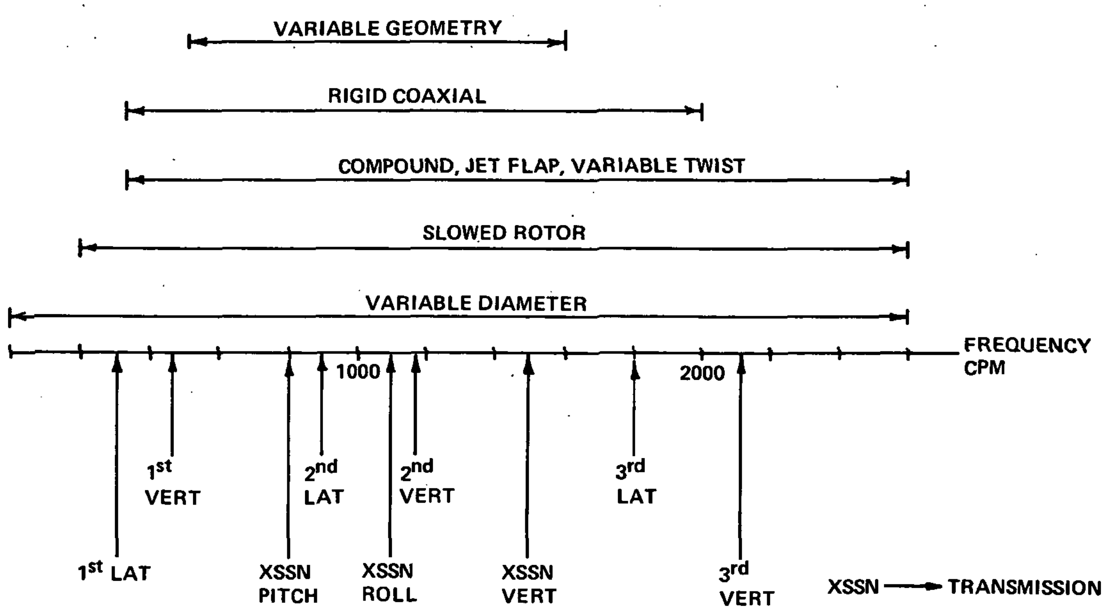
FIGURE 37

AIRFRAME DYNAMICS

Six possible advanced rotor systems have been considered for testing on the RSRA in addition to conventional compound rotors: the six-bladed variable geometry rotor, the variable diameter, rigid coaxial, jet flap, variable twist, and slowed rotors. Considering possible variations in blade number and tip speed, the wide band of principal blade passage frequencies that the RSRA might have to accept has been determined. These are illustrated in Figure 38. The general locations of airframe resonances typical of a single rotor helicopter of the RSRA size and configuration have also been determined. These modes shown are those of the first prototype S67 Blackhawk. The actual definition of modes for the RSRA must await detailed vehicle design.

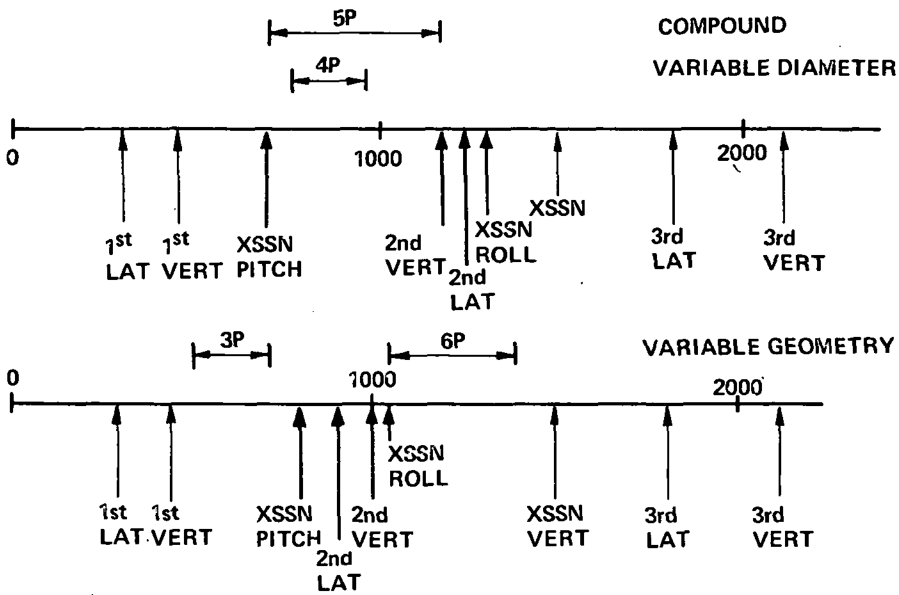
It would be impossible to design an airframe so that all modes of vibration will never be resonant with all vibratory excitation frequencies shown. However, the excitation bands produced by rotor configurations with the greatest near-term interest offer an environment in which the airframe structure can be dynamically tuned.

Consider the five-bladed compound rotor (forward speeds up to 300 kts), six-bladed variable geometry rotor, and the four-bladed variable diameter rotor, Figure 39 shows the bands of blade passage frequencies produced by each of these rotors. In each case, the bands are expanded by 10 percent on the high and low sides to provide acceptable resonance separation consistent with good design practice.



RSRA BLADE PASSAGE FREQUENCIES AND ANTICIPATED FUSELAGE MODES

FIGURE 38



LOCATION OF TUNED FUSELAGE MODES

FIGURE 39

In order to minimize the modal shifting required to accommodate these rotor systems, two tuned configurations are recommended. The first will accommodate the compound and variable diameter rotors. The second will accommodate the variable geometry rotor.

Figure 39 illustrates the locations of the tuned fuselage modes in proximity to the excitation frequency bands for the two configurations. The modes that must be controlled are the XSSN pitch, XSSN roll, second lateral, and second vertical bending modes. Experience indicates that these modes are uncoupled, and their locations are controlled by the stiffness of different portions of the airframe. The transmission pitch mode is controlled by the stiffness of the top of transmission support frames. The transmission roll mode is controlled by the stiffness of the sides of these frames. The second lateral and second vertical bending modes are controlled by the lateral and vertical bending stiffness of the aft fuselage and tail cone, respectively. The basic vehicle will be designed to locate the modes at the lower of the two required positions. The frequencies of these modes can then be increased as required through the addition of material.

The feasibility of shifting the location of fuselage modes has been demonstrated during full-scale ground tests at Sikorsky Aircraft. In addition, three-, five-, and six-bladed rotors have been successfully flight tested on a single aircraft, the S-61F (NH-3A) high speed research aircraft.

This airframe tuning provides the capability of testing the compound rotor, the variable diameter rotor, the variable geometry rotor, and any other rotor system whose primary excitation frequencies fall within the bands produced by these rotors.

THE OPTIONAL ROTOR BALANCE/VIBRATION SUPPRESSION SYSTEM

If RSRA is to accommodate certain unusual rotors which operate over a wide range of blade passage frequencies, such as the slowed rotor, an active vibration suppression system is required to avoid rotor/airframe dynamic resonances. This will provide tuning for other rotors whose frequencies fall beyond the bands available with the tuned airframe.

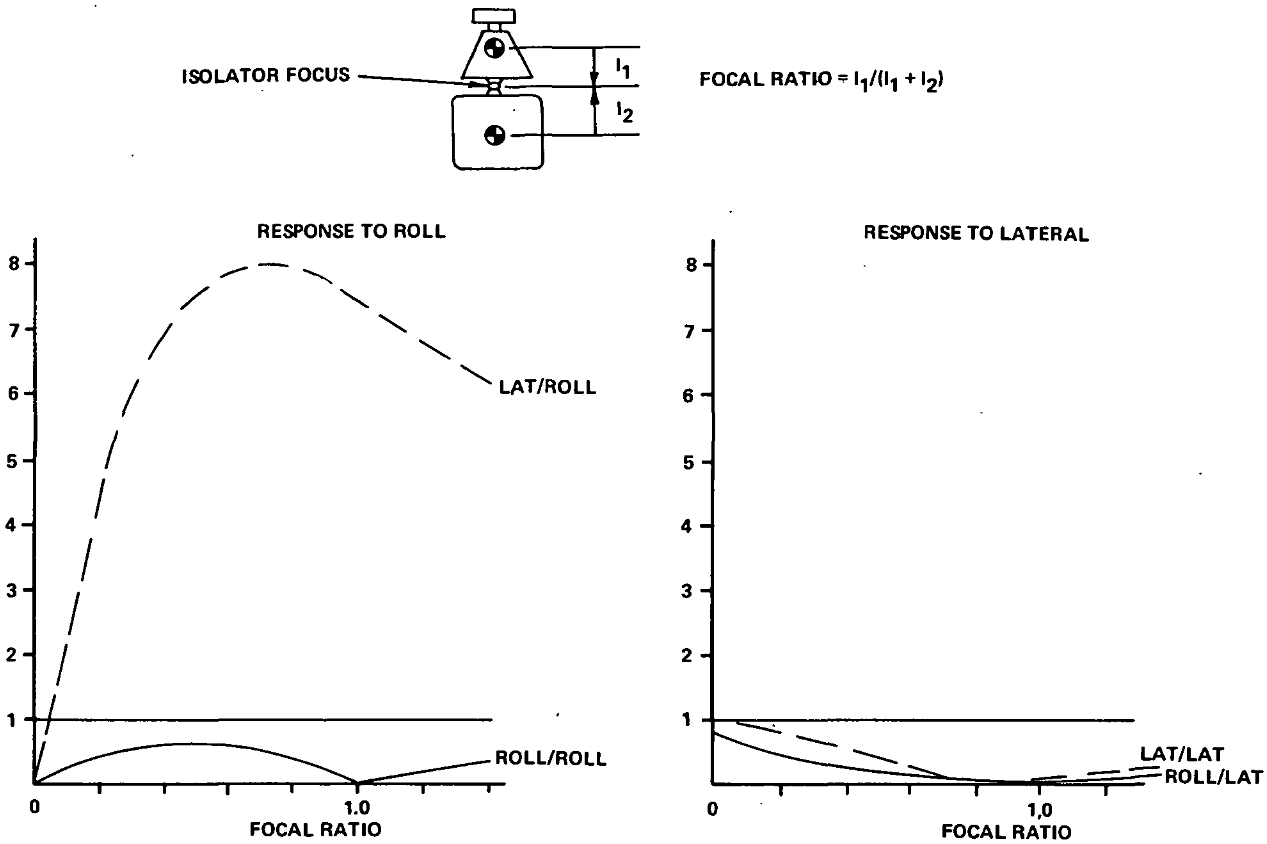
Active transmission isolation can provide all the wide band tuning characteristics required for a completely universal RSRA. Static and transient displacements would be actively controlled. Spring rates can be made as low as required to provide wide band isolation. It must be noted that the term isolation in this context defines a method of vibration suppression. An isolation system for RSRA is not intended to totally eliminate vibration nor can such a system be designed from a practical standpoint.

Passive isolation systems are limited in that there is a practical lower limit to their flexibility due to control system and engine shaft displacement limits, thus requiring the use of stops. One reason wide band passive isolation does not appear practical for RSRA is that the spectrum of steady rotor forces would tend to bottom the isolation system too often. Other disadvantages are that an auxiliary yaw restraint mechanism is required, yaw isolation cannot be provided except for coaxial rotors, and more significantly, simultaneous inplane force and moment isolation cannot be achieved.

Fixed system vibration absorbers, although tuneable, provide only local suppression and cannot prevent possible damage to structure or instrumentation due to high vibration levels in other areas. Anti-Resonant vibration isolation systems are tuned to a single frequency and thus cannot provide wide band RSRA vibration suppression.

Existing passive transmission/rotor isolation concepts cannot provide simultaneous isolation of forces and moments. Passive systems for inplane forces and moments can be ideally treated as two bodies connected by a hinge at the isolator focus (Figure 40). The focal ratio is defined as $\ell_1 / (\ell_1 + \ell_2)$ where ℓ_1 is measured from the upper body center of gravity (including the rotor effective mass) to the focus, and ℓ_2 is measured from the lower body center of gravity to the focus.

Typical ideal system transmissibilities to N/Rev roll and lateral excitations are illustrated in Figure 40. A transmissibility of 1.0 corresponds to the unisolated rigid body response. A focal ratio of zero corresponds to the focus at the upper body center of gravity, a focal ratio of 1.0 puts the focus at the lower body center of gravity.



TYPICAL FOCUSED ISOLATION TRANSMISSIBILITY CHARACTERISTICS
FIGURE 40

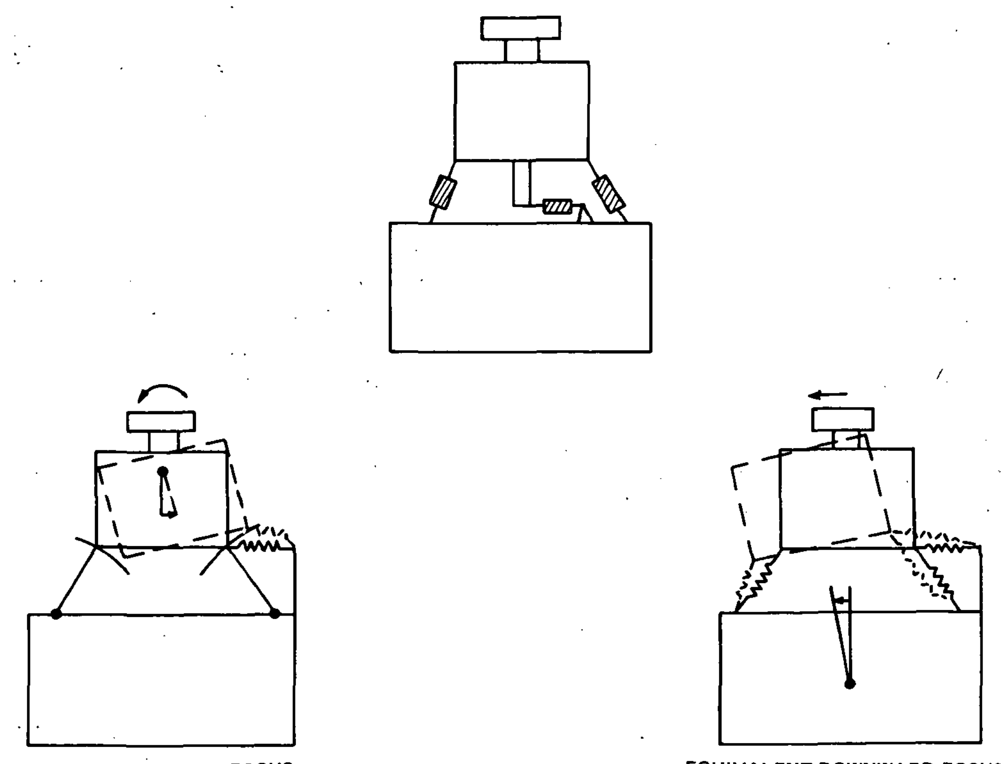
For a rotor with inplane vibratory force excitation and no moment, such as the teetering rotor, the system would be focused at or below the lower body center of gravity. For a vehicle with predominant vibratory moment excitation the system would be focused at the upper body center of gravity. However, this is not the general case. Most rotors produce some combination of force and moment excitation. From Figure 40 it can be seen that there is no single focus position that will isolate for both forces and moments simultaneously. In fact, a significant moment amplification (in this case 8:1) can be encountered if the system is not focused properly. This amplification can approach 40:1 in the pitch mode and thus can become significant even for articulated rotors.

CONCEPT OF UNIVERSAL VIBRATION SUPPRESSION SYSTEM

A Universal Active Vibration Suppression system can provide simultaneous isolation to all forces and moments while limiting static and transient displacements. The system can also serve as a rotor balance by providing a defined load path for measuring steady vibratory and transient loads.

The proposed configuration of the Sikorsky Active Rotor Balance/Vibration Suppression system is illustrated in Figure 41 . Seven self contained hydro-pheumatic actuators (isolaters) are shown. Seven are selected in order to decouple the pitch and roll modes and thus provide independent focusing. The orientation of the inplane units has been selected so as to decouple the load path of yaw moment and longitudinal force and thus provide the maximum longitudinal force measurement accuracy. Circular tracks are provided so that focusing can be easily varied. This variation can be accomplished independently in pitch and roll.

The ability to simultanesouly isolate inplane force and moment excitation is produced by the addition of lateral flexibility at the intersection of the canted focusing struts. A two dimensional schematic of the system is illustrated in Figure 42. The canted isolators cannot react pure moment since they intersect. They act as rigid focusing struts under the application of moment excitation. Since the units are active springs they provide an inherent lateral flexibility to inplane forces at their point of intersection. If a lateral force is applied, the upper body translates. The elastic center can be lowered through the addition of a lateral spring and therefore the upper body will translate and roll under the application of a lateral force. The effective points of rotation under force excitation can be made to lie within or below the lower body.



EQUIVALENT UPWARD FOCUS FOR MOMENT

EQUIVALENT DOWNWARD FOCUS FOR FORCE

KINEMATICS OF UNIVERSAL VIBRATION SUPPRESSION CONCEPT

FIGURE 42

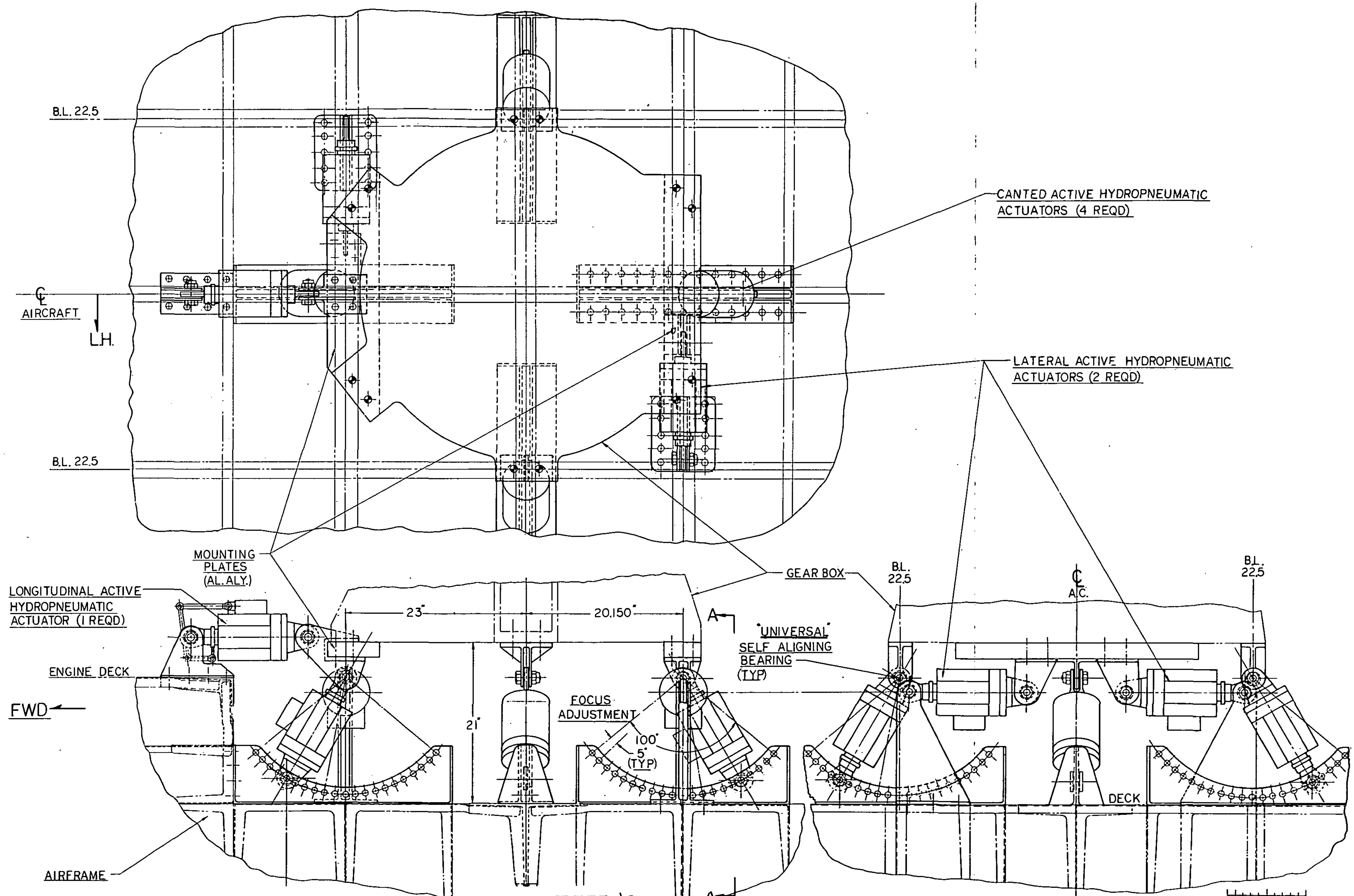
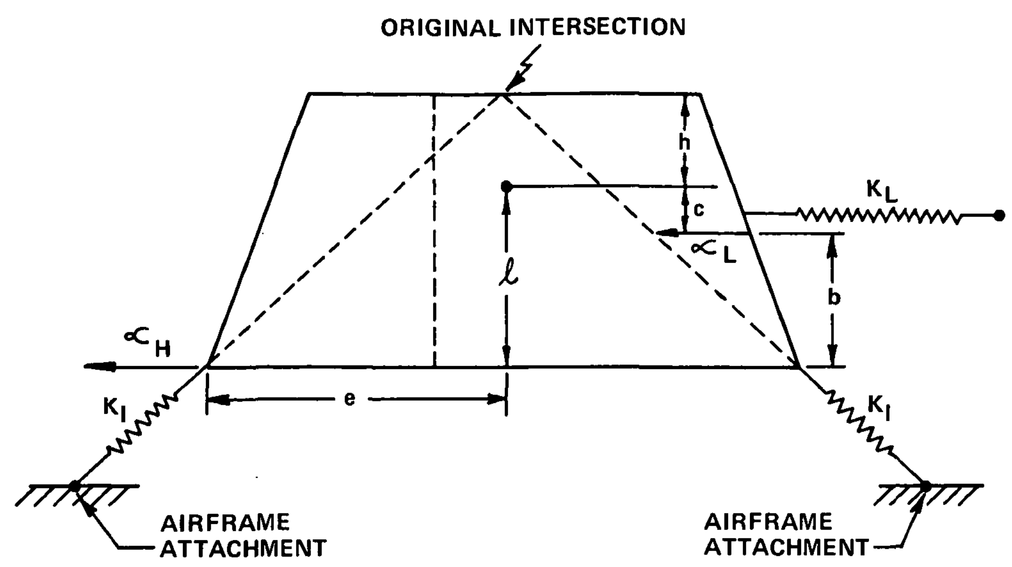


FIGURE 41

R.S.R.A. — GEARBOX — AIRFRAME — ACTIVE ROTOR BALANCE/VIBRATION SUPPRESSION SYS.



MATHEMATICAL MODEL OF UNIVERSAL
VIBRATION SUPPRESSION CONCEPT

FIGURE 43

The proposed concept is evaluated analytically by constructing a 3 degree of freedom mathematical model (Figure 43). The vertical degree of freedom is assumed to be uncoupled from lateral and roll. The model consists of two canted isolators to react vertical force and provide focusing in conjunction with an inplane isolator at the transmission attachment waterline. It is assumed that the three isolators provide the only load path from the rotor to the airframe.

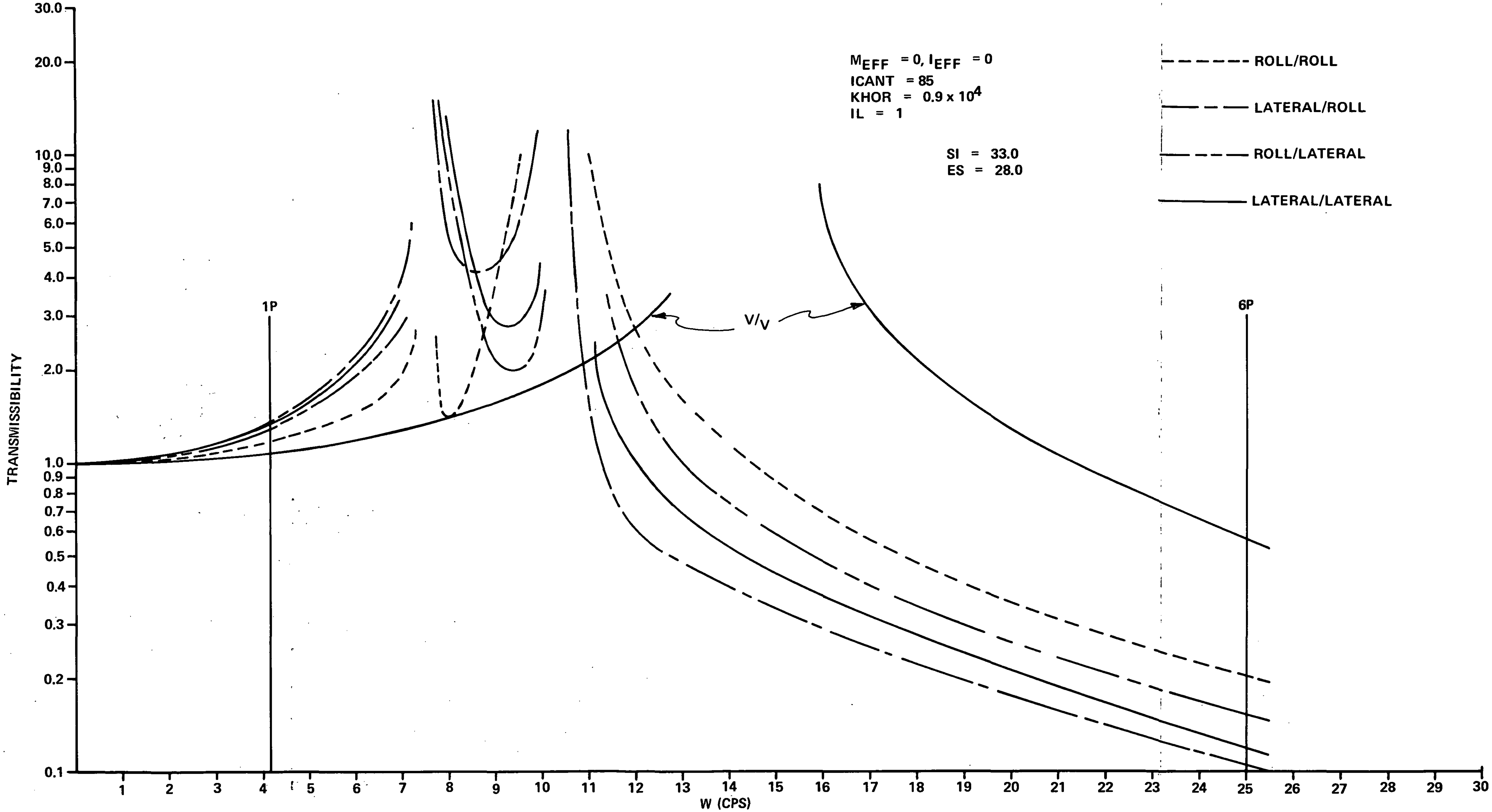
Focusing is provided by the canted isolators since they provide no torsional reaction about their point of intersection. There is a lateral spring rate at this focus point which results from the lateral component of the canted springs. In combination with the stiffness of the inplane isolator, required to react moment, an elastic center results which is located between the rigid focus point and the base of the transmission. The resulting system has two inplane modes; a relative rotation mode and a relative lateral mode. The introduction of the latter mode provides the mechanism for detuning the large amplification of inplane response to moments which can result in the rigid focused configuration. From the model illustrated in Figure 43, the following relations are developed:

$$\ell = \frac{bK_L + e\,(2\,K_I \cos\gamma \sin\gamma)}{K_L + 2\,K_I \cos^2\gamma}$$

$$K_{LAT} = K_L + 2K_I \cos^2\gamma$$

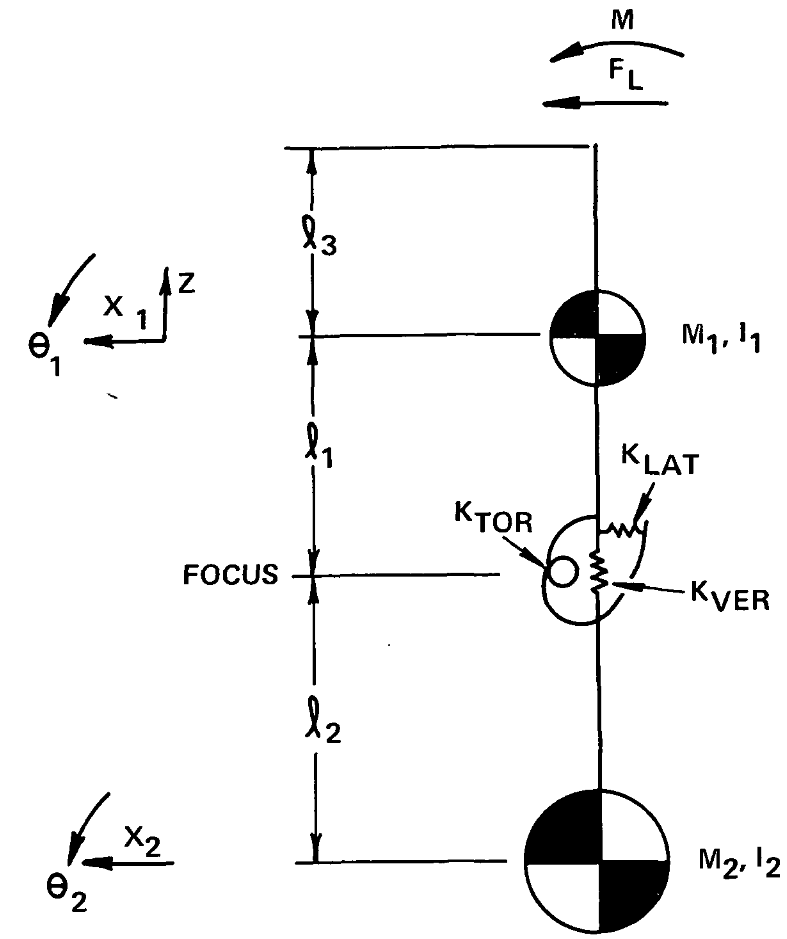
$$K_{TOR} = K_L C^2 + 2K_I \cos^2\gamma\,(h)^2$$

$$K_{VERT} = 2K_I \sin^2\gamma$$



TRANSMISSIBILITY VS EXCITATION FREQUENCY CASE 1

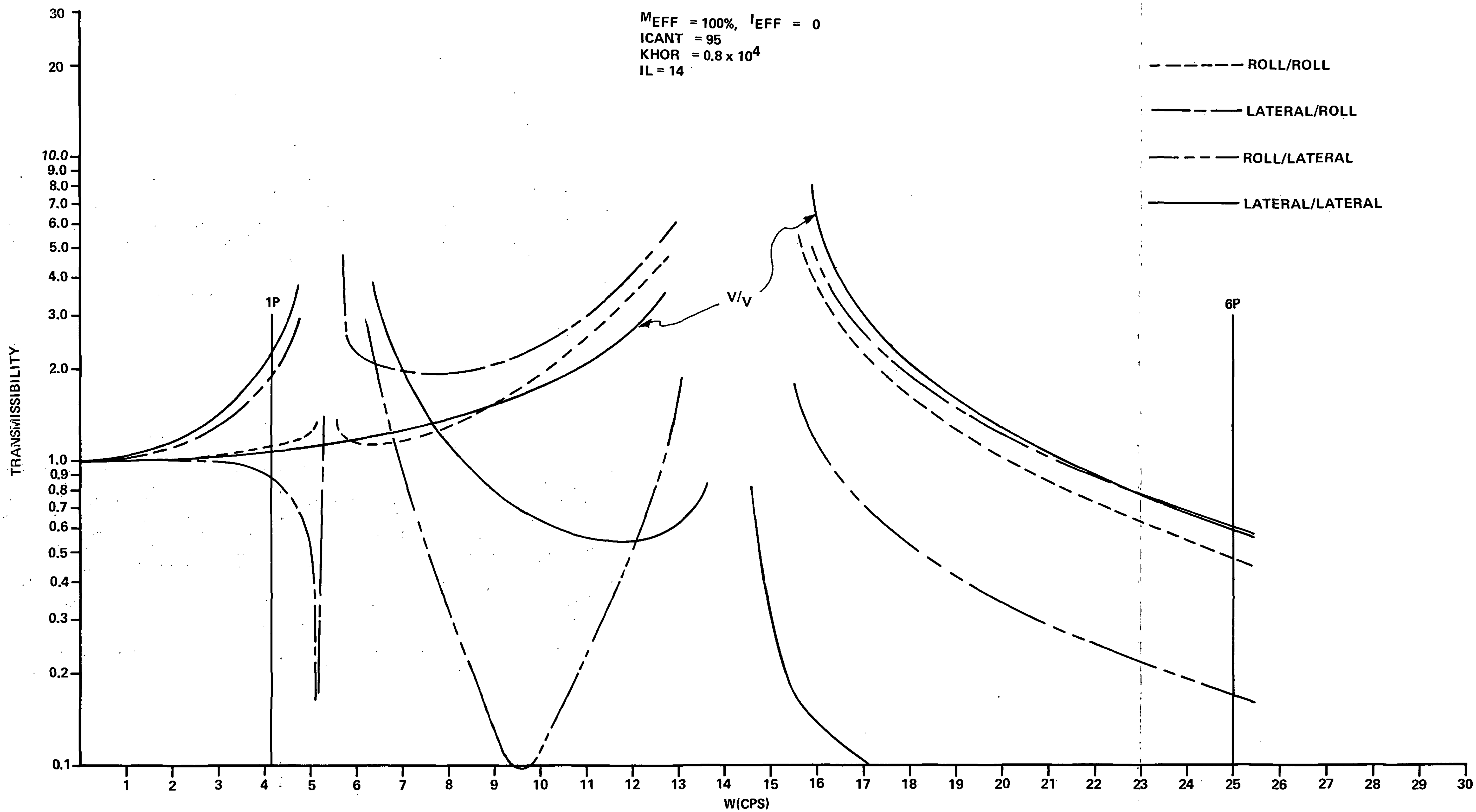
FIGURE 45



EQUIVALENT MATHEMATICAL MODEL
UNIVERSAL VIBRATION SUPPRESSION CONCEPT

FIGURE 44

The equivalent mathematical system is represented in Figure 44 where the three equivalent spring rates at the elastic center are assumed to connect the upper and lower bodies. The equations of motion were programmed on a UNIVAC 1108 to expedite evaluation of the concept. To simplify initial calculations the isolators are modeled as undamped springs. It is considered from previous experience (Reference 1) that this assumption does not significantly alter the prediction of isolation system capability. In each case analyzed, a Vertical Natural Frequency was selected for N/Rev isolation. The canted isolator spacing, cant angle and inplane isolator spring rate could then varied to establish simultaneous isolation to lateral forces and roll moments. A similar analysis can be performed in the longitudinal/pitch direction. All cases considered have isolator modes between 1/Rev and N/Rev.



TRANSMISSIBILITY VS EXCITATION FREQUENCY CASE 2

FIGURE 46

The following constraints are assumed in the preliminary analysis performed.

1) The fuselage attachment point of the canted isolators is fixed

2) The vertical location of the inplane isolator is fixed

3) The cant angle γ is limited to the following: $0 < \gamma < 90^{\circ}$.

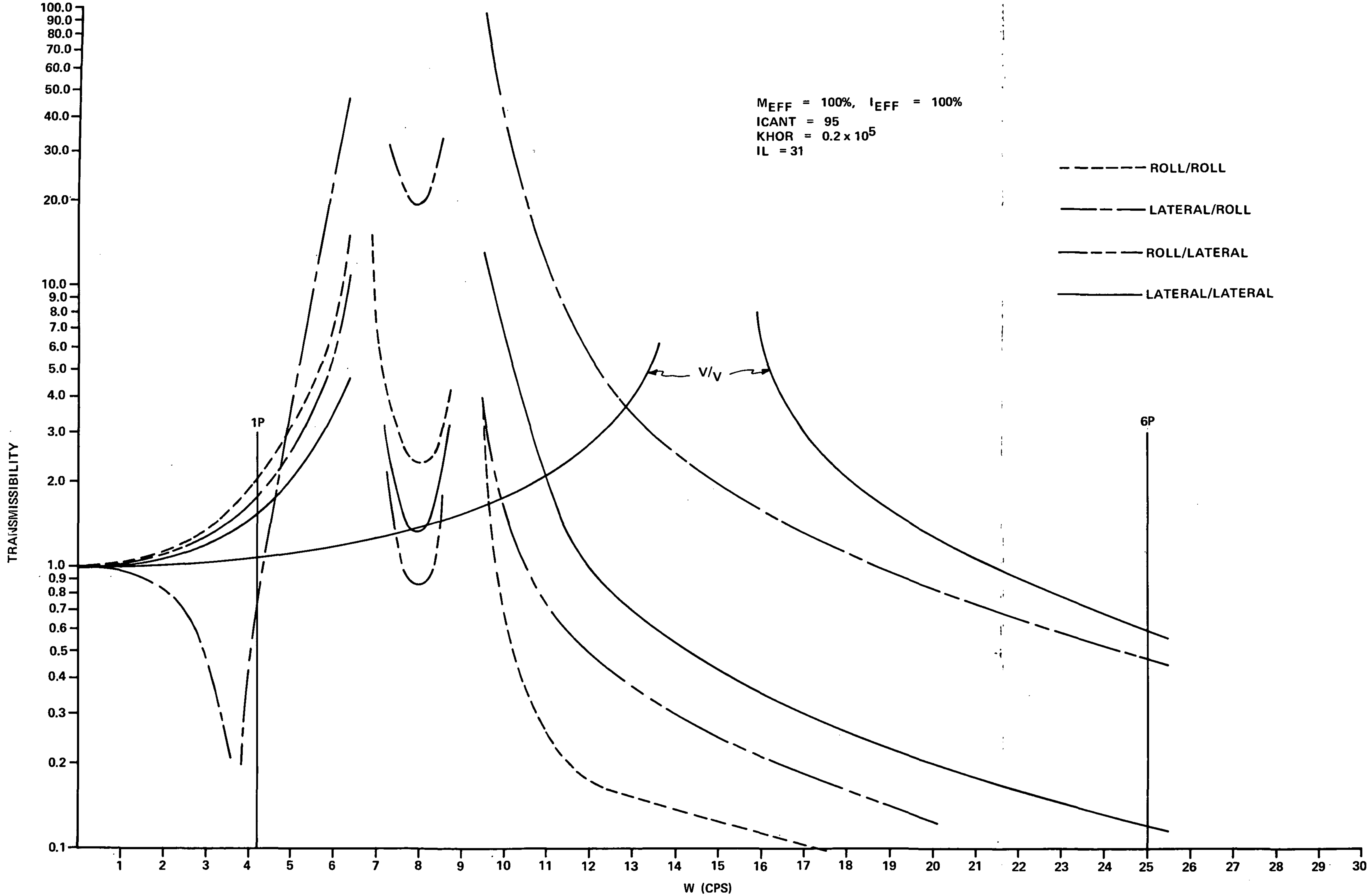
The rigid body inertia properties and C.G. locations of the various components of the preliminary RSRA used in the substantiation analysis is shown in Table XII.

The analysis was performed for a six bladed semi-rigid rotor configuration. Past experience indicates that the impedance of the rotor, in this instance the effective mass and roll inertia of the rotor, will vary depending upon the location of blade modal frequencies and their associated mode shapes. In order to include these parameters in the evaluation of the proposed system, four combinations of these parameters are considered.

	Effective Mass % of Rigid	Effective Inertia % of Rigid
1)	0	0
2)	100	0
3)	100	100
4)	0	100

In each case, the vertical natural frequency is set at 900 cpm. This establishes the required vertical spring rate. The cant angle is then set and the horizontal spring rate varied parametrically. The results of the analyses of the four cases specified above are shown in Figures 45-48. Illustrated are the variation of transmissibilities as a function of excitation frequency. The location of the three isolator modes are clearly defined in each case. In Figures 46-48 an apparent anti-resonant point appears in the lateral to roll transmissibility. This results from the combined lateral response of the two inplane isolator modes producing a forced response mode at the lower body center of gravity at the particular frequency.

It can be seen from each of these figures that simultaneous isolation to all three excitations can be achieved through the utilization of this concept.



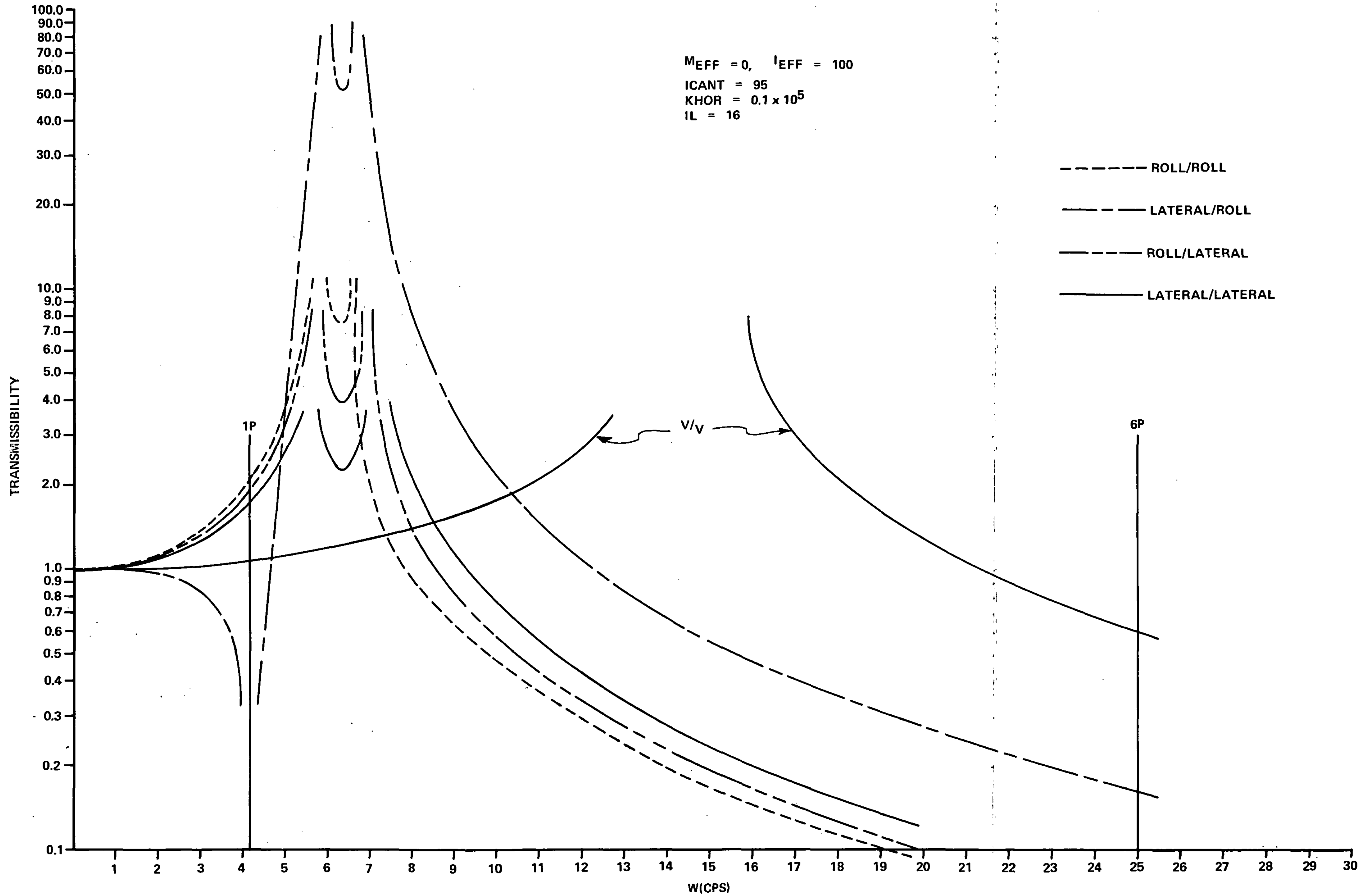
TRANSMISSIBILITY VS EXCITATION FREQUENCY CASE 3

FIGURE 47

COMPONENT	C.G. LOCATION (INCHES BELOW M.R. C.G.)	WEIGHT (LBS)	Ixx (SLUGS - FT^2)	Iyy (SLUGS - FT^2)	Izz (SLUGS - FT^2)
ROTOR	0	740	3000	3000	6000
HUB	0	900	30	30	60
TRANSMISSION	48	2000	200	200	100
AIRFRAME	100	21400	25000	125000	140000

PRELIMINARY RSRA INERTIA DATA

TABLE XII



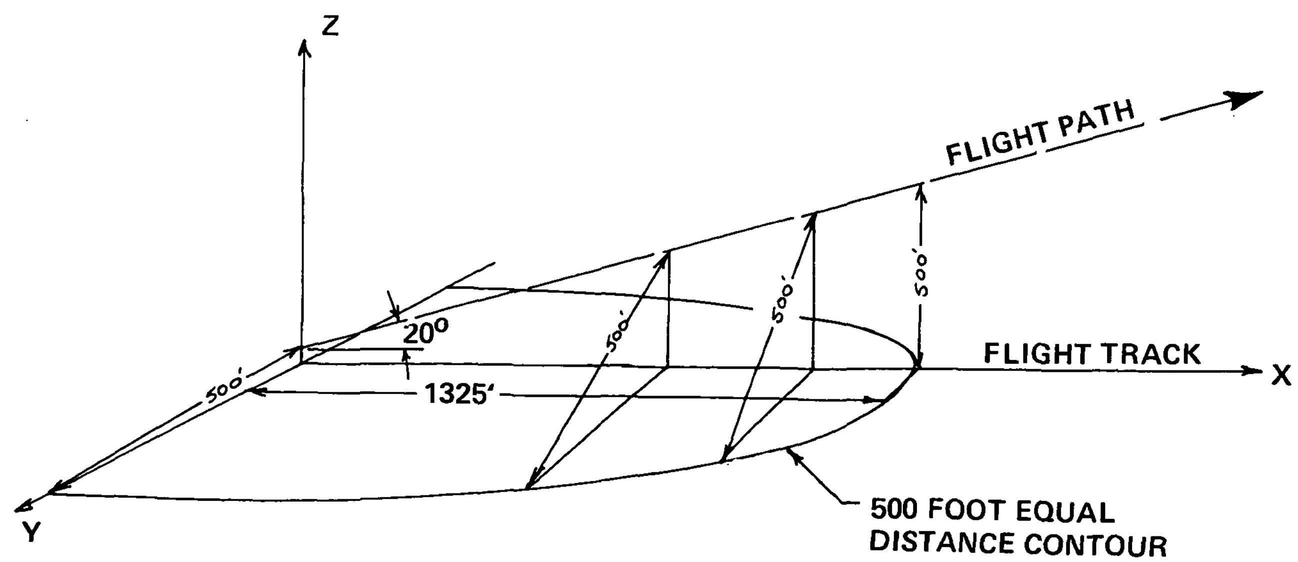
TRANSMISSIBILITY VS EXCITATION FREQUENCY CASE 4

FIGURE 48

AIRCRAFT EXTERNAL NOISE

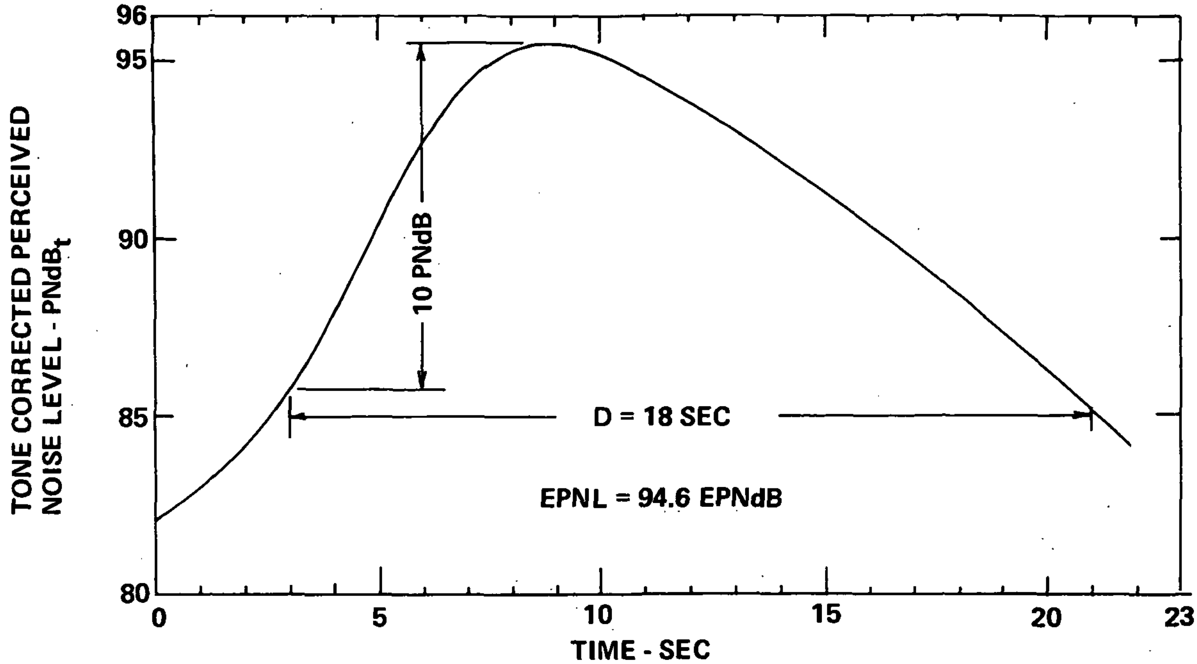
Aircraft Takeoff Noise

The external noise characteristics of the aircraft were calculated in order to verify that 95 EPNdB at a 500-foot sideline point would not be exceeded during takeoff and to determine if engine and fan noise are sufficiently low to allow a valid acoustic assessment of the main rotor noise. The calculated maximum Effective Perceived Noise Level on a 500-foot equal distance ground contour during take-off is 94.6 EPNdB, slightly below the 95 EPNdB criteria. This contour is a line on the ground any point of which is 500 feet from the aircraft at the closest point of approach. Figure 49 shows the contour. Take-off noise was evaluated at points on this contour rather than at points on a 500 foot sideline, since the equal distance contour represents the more stringent requirement. Points on the 500-foot sideline are more than 500 feet from the aircraft at its closest point of approach because of the increasing altitude during the takeoff climb-out. During Parts I and II of the study, it was determined that the maximum EPNL occurred at the contour point which is 900 feet from the takeoff point and 354 feet to the side of the flight track. Figure 50 presents the PNLT time history at this point during takeoff (assumed to be a 20 degree climbout at 50 knots from a 50-foot hover) from which the value of 94.6 EPNdB was computed.



500 FOOT EQUAL DISTANCE GROUND CONTOUR
FOR 20° CLIMBOUT FROM A 50 FOOT HOVER

FIGURE 49



CALCULATED PNLT TIME HISTORY AT 500 FOOT EQUAL DISTANCE GROUND CONTOUR POINT X = 900' Y = 354' DURING RSRA 20° CLIMBOUT FROM A50 FOOT HOVER

FIGURE 50

Component Noise Levels

Main Rotor

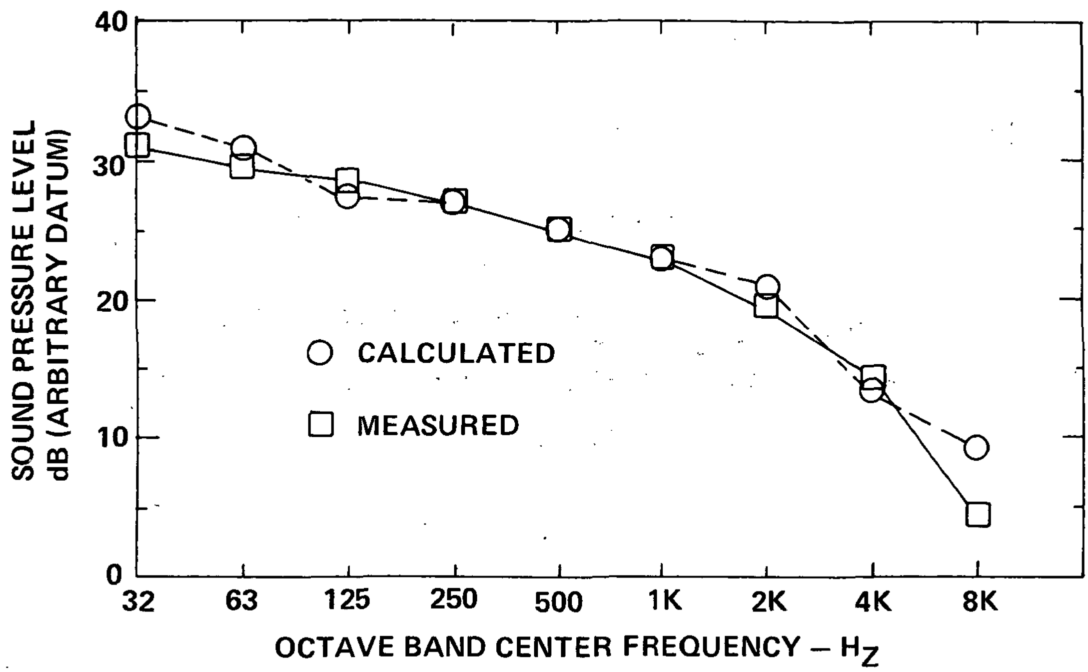
The noise signature of the baseline main rotor was calculated using a combination of the simplified rotational noise calculation procedure discussed briefly by Lowson and Ollerhead (Reference 2) and the broadband noise calculation method presented by Schlegel, King, and Mull (Reference 3). The Lowson/ Ollerhead procedure is a closed form Bessel Function type solution to the moving acoustic dipole radiation equation and differs from the classical Gutin (Reference 4) solution in that harmonics of the unsteady blade airloading are used in addition to the steady loads. In order to simplify the calculation, the airloading harmonics are calculated from the steady by the equation:

$$L_\lambda = L_o \, l_e \, \lambda^{-k}$$

where L_λ is the amplitude of the λ^{th} loading harmonic, L_o is the steady loading amplitude, l_c is a correlation length, and k is an arbitrary constant dependent on rotor system geometry. Studies performed at Sikorsky (Reference 5) have established methods by which the value of k and be estimated for a given rotor design.

The rotational noise procedure has been combined with the broadband noise calculation method developed in Reference 3 resulting in a program which calculates the complete rotor noise spectrum. This program has been found to correlate very well with the measured rotor noise as demonstrated by Figure 51.

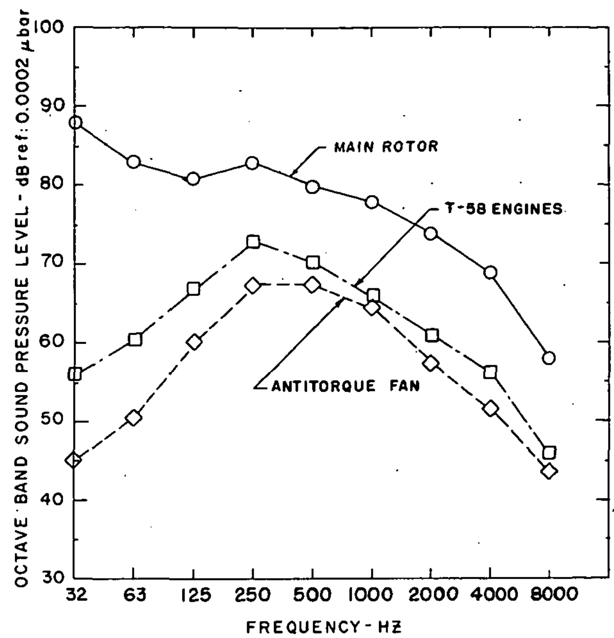
At some of the field points examined rotor noise can indeed be evaluated separately from the T-58 engines and anti-torque fan (Figure 52), however as shown in Figure 53, the shaft engine noise does dominate portions of the frequency spectrum near the front of the aircraft. It may become necessary to design engine silencing treatment for the T-58 engines if rotor noise evaluation in this area is to be investigated.



COMPARISON OF MEASURED AND CALCULATED MAIN ROTOR NOISE LEVELS
FIGURE 51

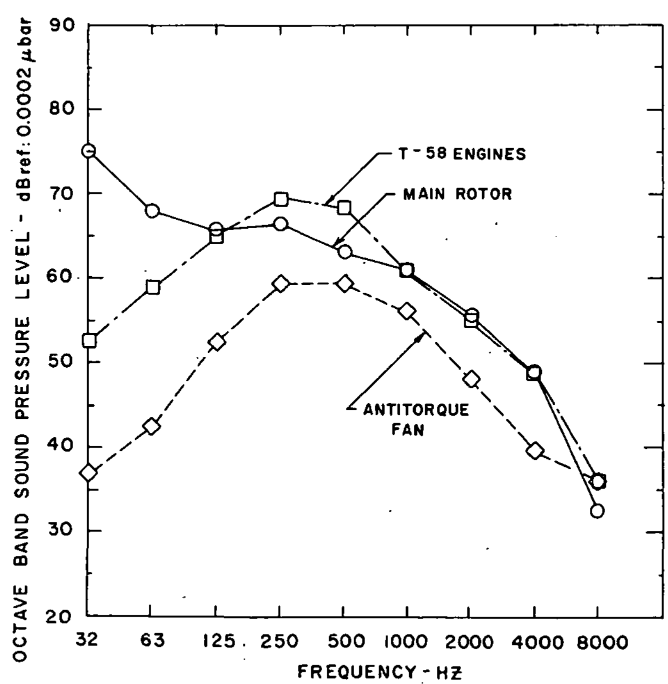
Anti-torque Fan Noise

Noise from the anti-torque tail fan was estimated by Hamilton Standard, developers of the fan. The levels as shown in Figures 53 and 54 are not as high as might have been expected, due principally to the low number of blades (7) and the relatively low tip speed (726 fps) design. The spectrum shape was estimated from preliminary model fan data and while preliminary appears to be a reasonable approximation and the levels are not expected to change significantly. Even though the fan may partially mask rotor noise at some locations (See Figure 53) no silencing treatment is presently planned. It may be necessary to further quiet the tail fan at a later date, depending upon the actual acoustic characteristics of the fan.



RSRA COMPONENT NOISE LEVELS 500 FEET
TO SIDE OF AIRCRAFT DURING TAKE-OFF

FIGURE 52



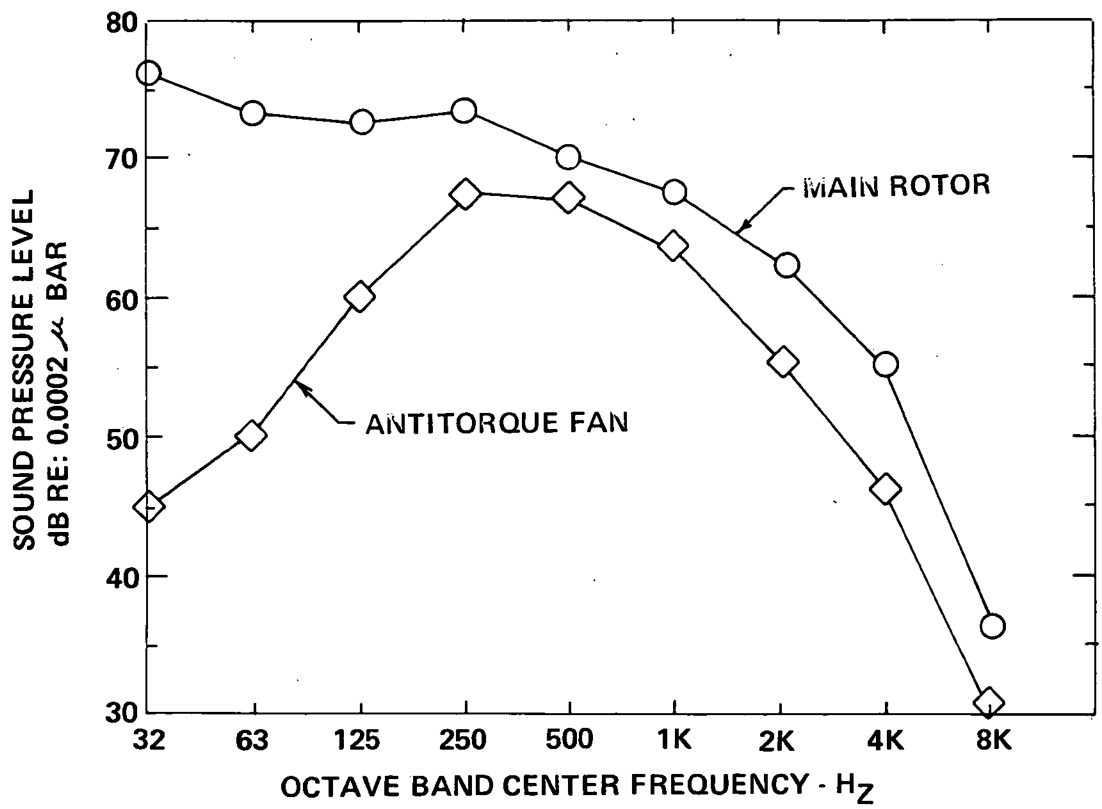
RSRA COMPONENT NOISE LEVEL 1000 FEET
FROM AIRCRAFT 20° OFF NOSE

FIGURE 53

Engine Noise

The GE-T58-16 engine noise was calculated from Sound Power Levels and directivity indicies presented in the GE T58 engine installation manual. The levels are corrected to account for atmospheric absorption (as are all the calculated component noise levels). As Figure 53 shows, the broadband noise components exceed main rotor noise levels near the front of the aircraft, thus it will not be possible to assess the rotor noise at points in this region. The engine noise does not, however, cause the vehicle noise to exceed the 95 EPNdB criteria on takeoff and silencing installations can be designed at a later date, if necessary, to further evaluate rotor noise.

TF-34 cruise fan noise is not a factor during takeoff and landings since the fans will be operating at ground idle conditions during normal takeoff. Their noise may interfere with rotor noise evaluation during high speed cruise. Silencing this engine should not be difficult because of the technology developed during the NASA Quiet Engine Program which used the TF-34 as the base engine.



COMPARISON OF MAIN ROTOR AND FAN NOISE LEVELS 1000 FEET FROM AIRCRAFT 45° OFF TAIL

FIGURE 54

AIRCRAFT RELIABILITY

Sikorsky reliability engineers have assisted in the conceptual design of the RSRA, providing their inputs to the various design tradeoff studies. Comments on RSRA reliability are as follows:

Rotor System

Experience with the S-61 rotor and blade systems is extensive on SH-3A, CH-3C, HH-3E, S-61N, SH-3D, S-67 and S61 Commercial configurations performing varied missions including air rescue, anti-submarine warfare, air support, cargo, commercial passenger carriers, and Apollo recovery. Historical data and established reliability values verify the high reliability of these assemblies.

Wing Group

The small RSRA wing does not present significant reliability problems. The large wing with all the added controllable surfaces will require a detailed reliability analysis of each control relative to failure modes and redundancy. This system is similar to conventional fixed wing aircraft, and no unusual problems are anticipated.

Anti-Torque System

Reliability trade-off analyses was part of early studies of several fan designs. Full failure mode and effect analysis will be required and reliability values on this portion of the aircraft system whould be included in RSRA detail design, construction and testing.

Tail Surfaces

These assemblies should pose no problem in defining reliability criteria. Reliability of fixed tail surfaces on S-61 helicopters has been excellent. The stabilator will require a reliability study.

Body Group

Reliability studies completed for S-61 series helicopter indicates no unusual problems in the basic airframe. Detailed analysis will have to be done during the aircraft design phase on the additional instrumentation, controls, and crew escape system.

Alighting Gear

Sikorsky has been designing and building retractable helicopter landing gear longer than any other helicopter manufacturer with experience beginning on production S-56 helicopters in the mid 1950's. Most Sikorsky helicopters designed and produced from then on have had retraction or kneeing alighting gear systems. Historical and reliability criteria is established, an no unusual problems are anticipated.

Flight Controls

Reliability failure mode and effect analysis and trade-off studies are most important in evaluating the flight control systems, and complete analyses of all control systems will be required during aircraft preliminary design. The S-61 control systems are well proven, however, the additions of fixed wing controls, fly-by-wire and computer increases the complexity of the system requiring greater emphasis on reliability analysis.

Drive System

The roller gearbox development included extensive detailed reliability analysis and is expected to be fully matured and have proven reliability. Tail drive shaft is standard with extensive historical data to prove high reliability. The tail gearbox will be analyzed with the anti-torque fan.

Onboard Data System

The importance of this system to the mission of the RSRA justifies a reliability program during aircraft design with emphasis on redundancy. The design is straightforward and no unusual reliability problems are anticipated if reliability is addressed from the start of the program.

Hydraulics

Current S-61 hydraulic systems have been purged of reliability sensitive parts and are proven systems. No additional problems are anticipated for the RSRA.

SAFETY REVIEW

In addition to the reliability reviews, safety engineers also reviewed the RSRA aircraft conceptual design to provide their inputs. Their comments are as follows:

The RSRA, its mission being primarily rotor research, faces new design requirements which introduce possible problem/risk areas over and above those which normally appear in the commercial or military helicopter designed for everyday, utilitarian use. While such basic things as main rotor/fuselage clearance, tail rotor/ground protection in flares, and the protection and retention of fuel in mishaps are still considerations, there is an added need for an immediate and positive in-flight escape mechanism, for more complex rotor and wing controls, and for added data measuring systems. Ground adjustable rotor mast tilt, inflight wing incidence and an all or partial fly-by-wire flight control system are other features required for research purposes which add possible problem/risk areas. The effect of these and other such design features on the safety of the RSRA has been carefully evaluated and treated in the design. One favorable point to be noted is that the aircraft will normally be flown by experienced test personnel in accordance with well thought-out (particularly from the safety viewpoint) test plans, rather than by the average military or civil helicopter pilot on a variety of missions.

Rotor System

The basic five-bladed S-61 rotor head, with over 1,250,000 hours of military and civil service, is used in the RSRA. Blade folding is unnecessary and the added complexities of this system are avoided. The RSRA rotor head is modified to the S67 rotor head configuration used on the Sikorsky "Blackhawk," whose rotor head controls have been modified to reduce any tendancy towards pitch-lag instability. This modification has effectively eliminated or reduced to only a slight degree any pitch-lag instability. A blade severance system is added as part of the emergency in-flight escape system (discussed later) , to provide optimum safety for the crew in case of uncontrollable emergencies during the rotor research.

Anti-Torque System

A common problem/risk area in anti-torque systems has been the susceptibility to damage during steep flares or by foreign objects such as hatch covers, flying debris, etc. The use of the variable pitch yaw fan greatly reduces the chance of damage due to foreign objects. In addition, the location of the fan in the tail surface, plus the aft location of the tail wheel, provides a maximum amount of protection from the ground in steep flares which might be encountered in autorotation. The tail rotor drive shaft is the improved large diameter design used successfully on the S61F type helicopter.

Flight Controls and Hydraulics

The fly-by-wire flight control system for control of the RSRA by the pilot has a mechanical back-up control system for the copilot. The rotor system servos have been proven by many hours and over 12 years of service.

Five hydraulic systems power the rotor, wing, and anti-torque flight control systems and the utility or test systems (drag flaps, fan louver, wing tilt and the landing gear and brakes). One hydraulic system powers only the primary servos; another powers only auxiliary servo, FAS, and one stage of the fan control. The rotor system, as in the S-61 and S-67, can be controlled by either the primary or auxiliary system alone. Dual systems also power the wing, and tail controls, the flap, aileron, vertical and horizontal stabilizers, and fan control. In addition, a 5.5 KVA constant speed generator is hydraulically driven by one hydraulic system to provide backup electrical power for research instrumentation at low rotor speeds. Cockpit warning systems warn of a low-pressure condition in any of the hydraulic systems. Pressure indicators are provided for 1, 2, 3, and 4 hydraulic systems. The fifth (rotor brake) system has no pressure indicator. Velocity valves in the No. 3 system isolate the flight control portions to prevent system loss due to a utility component malfunction. Also, a priority valve limits the flow to the No. 3 system utility component to provide sufficient flow for the flight control components. All four flight control hydraulic systems have individual, vented reservoirs with a remote cockpit fluid level indication.

Emergency Escape System

In case of an in-flight emergency requiring abondoning the RSRA, an emergency escape system is provided to sever the rotor blades, jettison the canopy, and separate each crew member from the aircraft. The Yankee system, well proven in several military aircraft, is used. The "tractor" principle of this system, pulling the crewman out of the aircraft, avoids the necessity of special provisions for the seat installations to withstand the firing shock, and the chance of compressive spinal injuries to the seat occupant. Blade severance has been tested previously in the industry; Sikorsky recently demonstrated its successful application to the S-61 rotor system.

AIRCRAFT WEIGHTS AND BALANCE

The group weights for the aircraft were derived where applicable from actual weights of existing components, by statistically derived parametric equations, by manufacturers specifications, by layouts, and by target weights of components currently under development in other programs. The weight breakdown by group for the three basic aircraft configurations is shown in Table XIII . The following is an explanation of the basis of estimating each group weight.

Rotor Group - The S-67 Blackhawk$_{TM}$ rotor system less bifilar absorber plus a blade severance system.

Wing Group - Estimated parametrically based on a gross wing area plus weight increments for wing tilt mechanism and wing load instrumentation. The helicopter configuration has no wing.

Tail Fan - Current fan target weight from the present U.S. Army/Sikorsky fan-in-fin study.

Tail Surfaces - Estimated parametrically for a 90 sq. ft. stabilator and a 50 sq. ft. vertical fin with a 15 sq. ft. rudder.

Body Group - Estimated parametrically based on a wetted area of 1110 sq. ft. plus weight increments for wing tilt, wing instrumentation, drag device, canopy separation, and ballast system. The helicopter configuration includes a wing notch fairing.

Alighting Gear - Estimated parametrically based on 8 feet per second sink speed and 120 knot landing speed.

Flight Controls - S-67 helicopter controls plus estimates for additional hydraulic power boost, force augmentation system, wing tilt, flap controls, aileron controls, stabilator controls, tail fan controls, rudder controls, drag device controls, and tail fan shutters based on analysis of system layouts and schematics. Wing-mounted controls are deleted for the helicopter configuration.

Rotor Propulsion - Manufacturer's weights are used for the two T58-GE-16 engines. Engine section and engine related items are derived from the S-67 with allowances for increased size and power.

Auxiliary Propulsion - Manufacturer's estimates are used for the two TF34-GE-100 engines. Engine section and engine related items are estimated as similar to the TF34 installation on the Lockheed S-3A. Auxiliary propulsion is deleted for the helicopter configuration.

Fuel System - Estimated parametrically for a bladder tank system having a usable capacity of 5000 lb of JP-4. Plumbing in the auxiliary propulsion pods is deleted for the helicopter configuration.

TABLE XIII
RSRA WEIGHT BREAKDOWN

ITEM	COMPOUND CONFIGURATION Aux. Propulsion Installed, Small Wing	"SIMULATION" CONFIGURATION Aux. Propulsion Installed, Large Wing	HELICOPTER CONFIGURATION No Auxiliary Propulsion No Wing
Rotor Group	2104	2104	2104
Wing Group	1125	2411	0
Tail Fan	360	360	360
Tail Surfaces	503	503	503
Body Group	3518	3518	3553
Alighting Gear	1098	1098	1098
Flight Controls	1627	1707	1577
Engine Section	989	989	216
Engines	3776	3776	886
Engine Related Items	346	346	240
Fuel System	304	304	286
Drive System	2476	2476	2476
Instrumentation	567	567	552
Hydraulic System	54	54	54
Electrical System	403	403	403
Avionics	260	260	260
Furnishings	284	284	266
Air Conditioning	136	136	136
Auxiliary Gear	30	30	30
Vibration Suppression	0	0	0
Contingency	599	599	599
Weight Empty	20559	21925	15599
Crew	400	400	400
Engine Oil	80	80	40
Unusable Fluids	40	40	35
Fuel	3313	3947 (Fuel and Payload)	2202
Payload	2000		2000
Gross Weight	26392	26392	20276

Drive System - The existing development version of the 3700 HP main gearbox and shaft is utilized. S-67 shafting, rotor brake, and oil cooling systems are retained. No intermediate gearbox is used. The tail gearbox is included with the fan weight.

Instrumentation - Basic flight and engine instruments are similar to the S-67 with the addition of auxiliary propulsion instruments. Included, in addition, is instrumentation for wing tilt, wing loads, auxiliary propulsion thrust, tail fan thrust, and rotor loads. The auxiliary propulsion thrust instrumentation is deleted for the helicopter configuration.

Hydraulic System - The weight of utility hydraulics, also used for some control devices, is estimated by analysis of layouts and schematics.

Electrical System - Similar to the S-61 commercial D.C. system. Estimated by component analysis.

Avionics - Includes VOR, DME, ILS, transponder, VHF, UHF, Intercom, and gyro compass capabilities. Estimated by component analysis.

Furnishings - Accommodations for personnel include YANKEE upward extraction systems for pilot and copilot and non-removable seat provisions for a third crewman in the cabin. Miscellaneous equipment is assumed the same as S-67. Emergency equipment includes fire extinguishing systems for all engines and two hand fire extinguishers. Part of the auxiliary propulsion fire extinguishing system is deleted for the helciopter configuration.

Air Conditioning - Assumed dual air conditioning system similar to the S-67.

Auxiliary Gear - A weight allowance is made for the potentially greater handling provisions for a research aircraft.

Weight Contingency - Assumed at 3% of all subsystem weights.

Balance

The balance characteristics of both the compound and helicopter version were estimated based on the following assumptions:

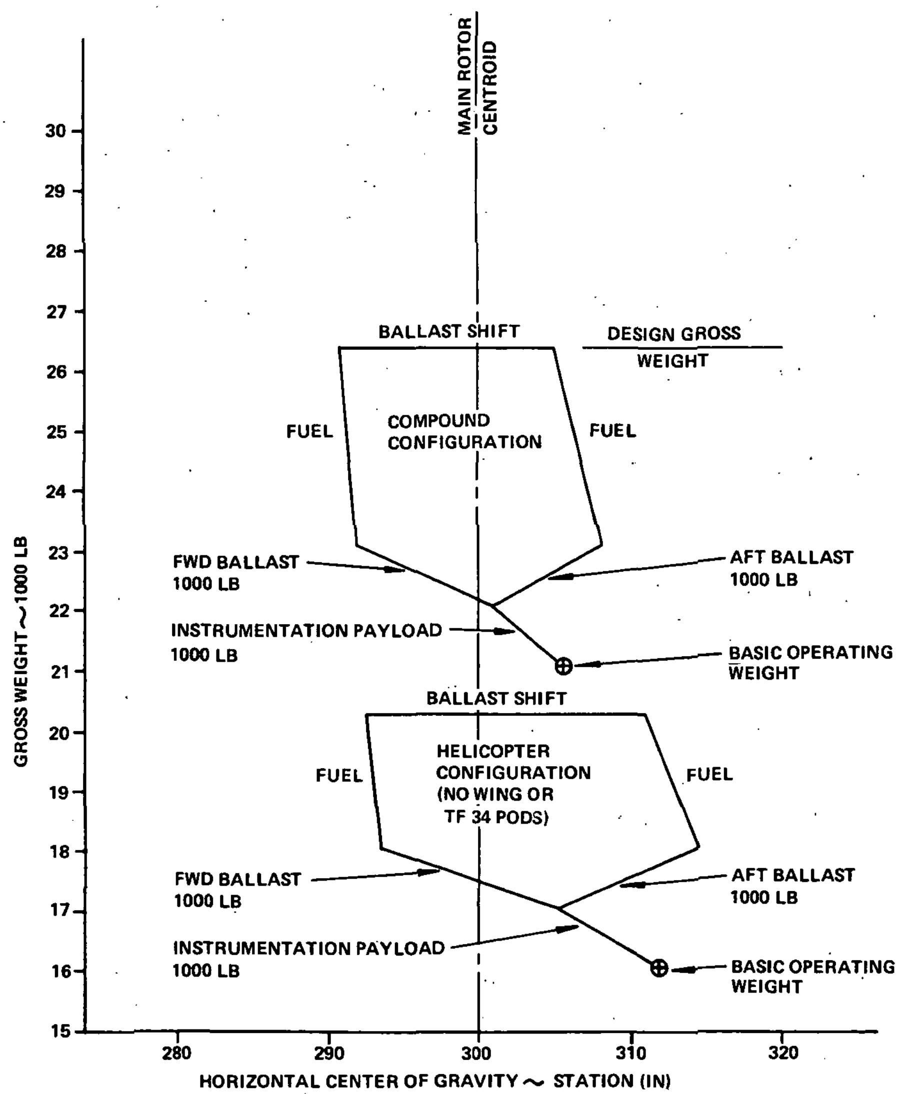
1. The average horizontal center of gravity of the mission payload is located at Station 200, 100 in. forward of the main rotor centroid.

2. 1000 lb of instrumentation payload is premanently installed on the aircraft.

The horizontal center of gravity of the compound configuration at its design gross weight of 26,392 lb is at station 294.8 or 5.2 in. forward of the main rotor centroid. At its mission gross weight of 20,276 lb, the helicopter configuration's horizontal center of gravity is at station 297.7 or 2.3 in. forward of the main rotor centroid.

Figure 55 shows horizontal center of gravity excursions for both configurations when 1000 lb of ballast is substituted for 1000 lb of removable payload. The ballast system provides more than 14 in. of center of gravity shift at design gross weight. This increases at lighter gross weight conditions such as reduced fuel or helicopter configuration, and decreases for overload weight conditions. Center of gravity variations with gross weight are as follows:

Gross Weight-Lb	Total CG Shift-In
18,000	20.9
22,000	17.1
26,000	14.5
30,000	12.5



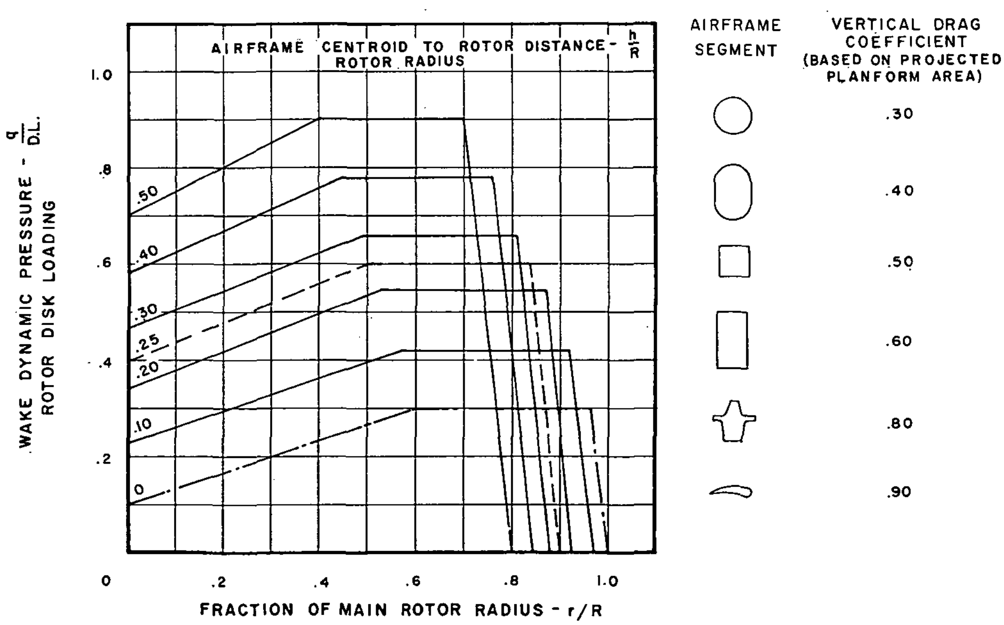
BALANCE CHARACTERISTICS
FIGURE 55

The performance of the RSRA aircraft was calculated for compliance with the Statement of Work requirements. Vertical drags were calculated using the NASA/Army method of the Statement of Work. Sikorsky methods indicated higher vertical drags and were used in hovering performance estimates. The equivalent parasite area for the aircraft has been estimated primarily using the NASA/Army method, as Sikorsky estimates indicate possible lower areas. Engine performances are manufacturers specifications with SFC's increased by five percent, and forward flight performance is executed using Sikorsky techniques which have been shown during the study to be a more conservative approach than that originally requested in the Statement of Work.

Vertical Drag

The complete aircraft vertical drag in hover with the large wing installed is equivalent to 6.72% of the gross weight or a net drag of 1771 pounds at the design gross weight of 26,392 pounds. The disk loading at this condition is 8.74 pounds per square foot.

The vertical drag was determined using the method of analysis outlined in Section 6.2.4.1(g) of the RSRA Contract Statement of Work. Figure 56 is reproduced from the Statement of Work with interpolated lines used in the analysis. The airframe was divided into 25 segments as shown in Figure 57. Each segment was assigned a vertical drag coefficient consistent with the figure. Segment planform area (A), distance of the segment centroid below the rotor (h/R), and distance of the centroid from the center of rotation (r/R) were obtained from the general arrangement drawing.



VERTICAL DRAG CALCULATION CHART

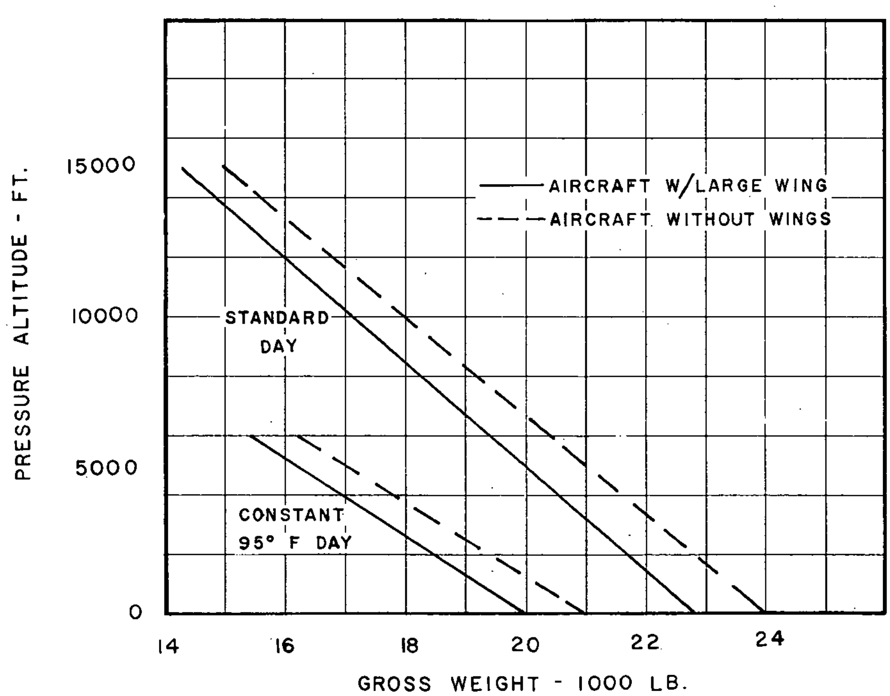
FIGURE 56

Details of the calculation are shown in Figure 57 . Figure 56 is entered at the proper h/R and r/R to read wake dynamic pressure over rotor disk loading for each airframe segment. This is multiplied by the disk loading of 8.74 pounds per square foot to obtain q. Drag on each segment is equal to q x area x C_D. The total vertical drag is the sum of the segment drags. The shaft is tilted forward 2 degrees and the coning angle is 3 degrees. The wing is at zero incidence with high lift devices retracted. The resulting value of 6.72% is applicable over a range of disk loadings. Previous work has shown that the method results in a constant value of vertical drag, expressed as a percentage, as long as the radius remains the same.

The vertical drag was also calculated by means of the Sikorsky standard procedure, Reference 6 . The value obtained for the complete aircraft is 7.72%. This method considers additional effects such as thrust recovery due to the presence of the airframe, variation in drag coefficients with thickness ratio, and dynamic pressure increases in areas such as the wing root where the body interferes with the normal wake distribution. Areas of particularly high vertical drag include the wing and the nose. The wing accounts for over half of the aircraft vertical drag.

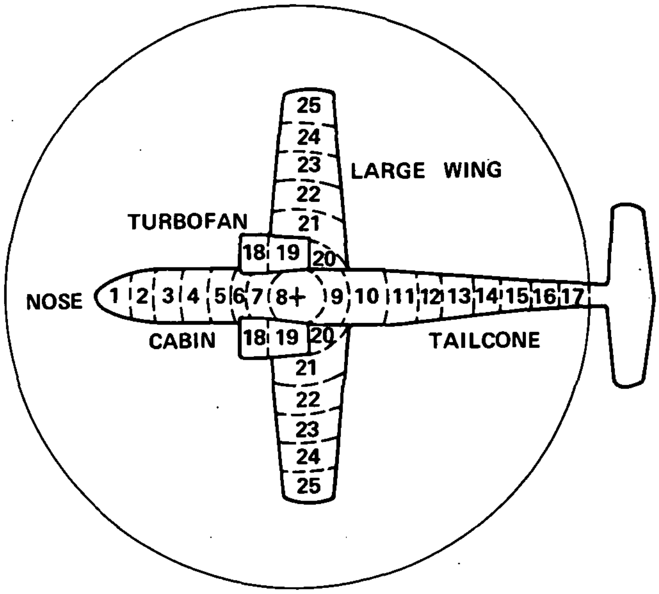
Hovering Performance

Out-of-ground effect hover capability of the RSRA is shown in Figure 58. Weight-Altitude-Temperature curves are presented for a standard day and a 95°F day.



RSRA HOVER CAPABILITY
TWO T58-GE-16 ENGINES @ MIL POWER
TURBOFANS REMOVED OGE

FIGURE 58



ELEMENT	r/R	h/R	q/D.L.	q (psf)	AREA (sq ft)	C_D	ELEMENT DRAG(LB)	COMPONENT
1	.702	.285	.643	5.63	13.3	.30	22.4	Nose
2	.594	.229	.572	5.03	17.5	.40	35.3	Cockpit
3	.503	.183	.515	4.50	17.65	.40	31.8	
4	.412	.175	.467	4.08	18.05	.50	36.9	Cabin
5	.317	.169	.425	3.72	19.6	.50	36.4	
6	.234	.067	.265	2.32	13.75	.60	19.2	
7	.15	.048	.213	1.87	22.45	.80	33.6	
8	0	.032	.140	1.23	30.2	.80	29.7	
9	.15	.046	.203	1.775	22.45	.60	23.9	
10	.272	.086	.303	2.655	24.6	.40	26.1	Tailcone
11	.393	.169	.450	3.94	15.3	.40	24.1	
12	.473	.194	.520	4.55	10.1	.40	18.5	
13	.55	.210	.557	4.88	13.1	.40	25.6	
14	.65	.229	.577	5.05	10.2	.40	20.7	
15	.75	.247	.597	5.23	8.65	.40	18.2	
16	.85	.266	.440	3.85	6.84	.40	10.6	
17	.95	.250	0	0	4.83	.40	0	
18	.282	.126	.357	3.21	29.0	.30	27.9	Turbofan (2)
19	.196	.132	.335	2.93	39.5	.30	34.6	
20	.15	.333	.567	4.97	32.68	.90	146	Large Wing (2)
21	.25	.336	.605	5.29	51.46	.90	245	
22	.35	.339	.647	5.67	49.32	.90	252	
23	.45	.339	.683	5.98	41.92	.90	226	
24	.55	.341	.707	6.18	36.26	.90	202	
25	.65	.341	.707	6.18	40.2	.90	224	

TOTAL DRAG (LB) 1771 LB
VERTICAL DRAG (% OF GW) 6.72%

FIGURE 57
VERTICAL DRAG BREAKDOWN

Capability has been evaluated for the aircraft with the turbofans removed. Power is provided by two T58-GE-16 engines at military rating. Capability is shown with and without the large wing.

Since the RSRA rotor system is nearly identical to that of the S-67, S-67 tether test data is used as a basis for hover performance calculation. The S-67 test data is shown in Figure 59 in non-dimensional form. These data are adjusted by 2.7% (the accepted value for S-67 vertical drag) to obtain thrust coefficients. The test data includes a -8° and a -4° linear twist; -3° twist performance is extrapolated. Making these corrections results in a non-dimensional C_T - C_P curve which is directly applicable to the RSRA. This curve is shown in Figure 60.

Power available is obtained from the manufacturer's specifications. Based on current fan-in-fin studies, a total hover efficiency of .78 is estimated.

A point on the Weight-Altitude-Temperature curve, Figure 58 , is obtained as follows. Main rotor power available is 78% of the shaft horsepower given in the manufacturer's specifications. Non-dimensionalize this by the power factor and enter Figure 60 at this C_{PM}. Read C_T and multiply by the thrust factor to obtain thrust. Reduce the thrust by the vertical drag to obtain gross weight.

With the turbofans removed, vertical drags are 7.64% with the wings on and 3.08% without the wings. These values were calculated by the Sikorsky standard procedure. As discussed in the section on vertical drag, this results in a more conservative approach than using the NASA vertical drag method.

Figure 59 lacks a trend with Mach number due to insufficient test data. The test results of the -4^{o} twist blade show no clear Mach number trend. The Mach number of .614 shown in Figure 60 is the 59°F case. Rather than attempt to estimate a 95°F trend, the approach has been to use this same line. Thus, any small errors which may result will be on the conservative side.

With hover mission fuel, the minimum operating weight of the aircraft (with wings and auxiliary propulsion removed) is 18276 lb. Thus, the aircraft can perform the desired mission on both the standard day and the 95°F day. With the wing on,aircraft gross weight is 20,782 lb excluding payload. The mission could therefore be performed with the wing on at standard day condition, but would have to be cut to 20 minutes, or reserves reduced, on the 95°F day.

Parasite Drag

The equivalent parasite flat plate area for the RSRA has been estimated by two methods, the Sikorsky standard procedure and the Sikorsky standard procedure with NASA/Army minimums imposed. The resulting total aircraft parasite areas with the small wing are as follows:

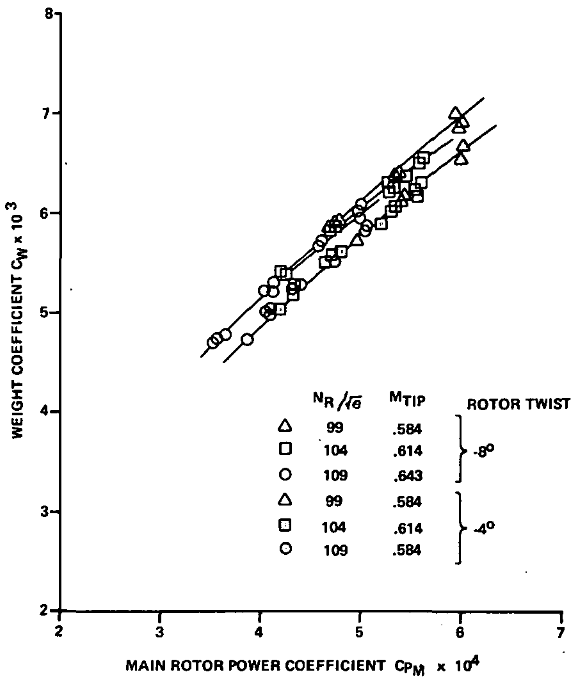
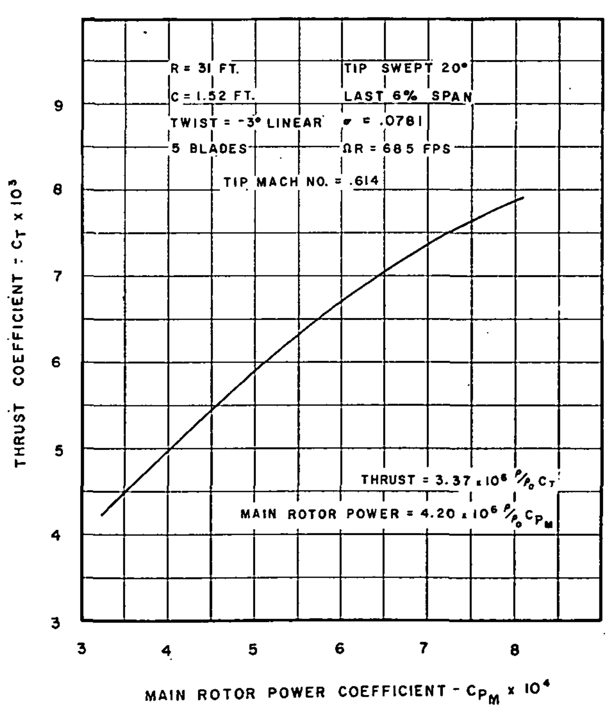


FIGURE 59



NON-DIMENSIONAL HOVER PERFORMANCE

FIGURE 60

Sikorsky Standard Procedure	20.6 ft^2
With NASA/Army Minimums	23.6 ft^2

The component breakdown for the latter method is shown in Figure 61. The following minimum drag formulas are included in the NASA/Army method as given in the Statement of Work.

$$f_{\text{wing, empennage, fuselage}} = .007 \times (\text{total wetted area})$$
$$f_{\text{hub and mast}} = .06\ (\text{main rotor horsepower})^{0.58}$$

These equations result in values of 10.37 square feet and 7.02 square feet respectively. In addition to these values, parasite areas were estimated for the T58-16 and the TF34-100 installations and an additional 1.11 square feet for protuberances and leakage.

FIGURE 61

PARASITE DRAG BREAKDOWN

Sikorsky Standard Procedure with NASA/Army Minimums

COMPONENT	PARASITE AREA - FT^2
Fuselage (Wetted Area = 891 ft^2)	6.31
Small Wing (Wetted Area = 281 ft^2)	1.97
Empennage (Wetted Area = 299 ft^2)	2.09
Hub and Mast	7.02
Turbofan Nacelles	3.75
Tail Fan (covered)	0.00
T58-16 Installation	1.30
Miscellaneous	1.11
Total	23.6

FORWARD FLIGHT

High speed forward flight performance has been evaluated at sea level standard and 9500 ft standard conditions. Thrust required, as a function of airspeed, is shown in Figure 62 with TF34-GE-100 estimated installed available thrust.

To avoid exceeding the critical tip Mach number of .94, the tip speed (ΩR) is reduced to 542 feet per second. The equivalent flat plate area (f) of the aircraft excluding wing induced drag is 23.6 sq. ft. The rotor shaft angle is +3°. Wing area is 184 sq. ft. The wing lift share is 80% of the gross weight or 21063 pounds. The rotor lift share is 20% or 5266 pounds.

The thrust or propulsive force is made up of basic airframe parasite drag, wing drag, and main rotor drag and H force.

At high speeds (250 - 300 knots) the majority of the propulsive force is needed to overcome parasite drag. This contribution is calculated as the product of the equivalent flate plate area (f) and the free stream dynamic pressure (q). That is:

$$\text{Parasite Drag} = qf = \tfrac{1}{2}\rho V^2 f$$

Another significant thrust requirement results from forces on the partially loaded main rotor. Rotor forces were calculated using Sikorsky's general rotor performance computer deck with skewed flow effects taken into account as approved by NASA/Army during the study (See Appendix B).

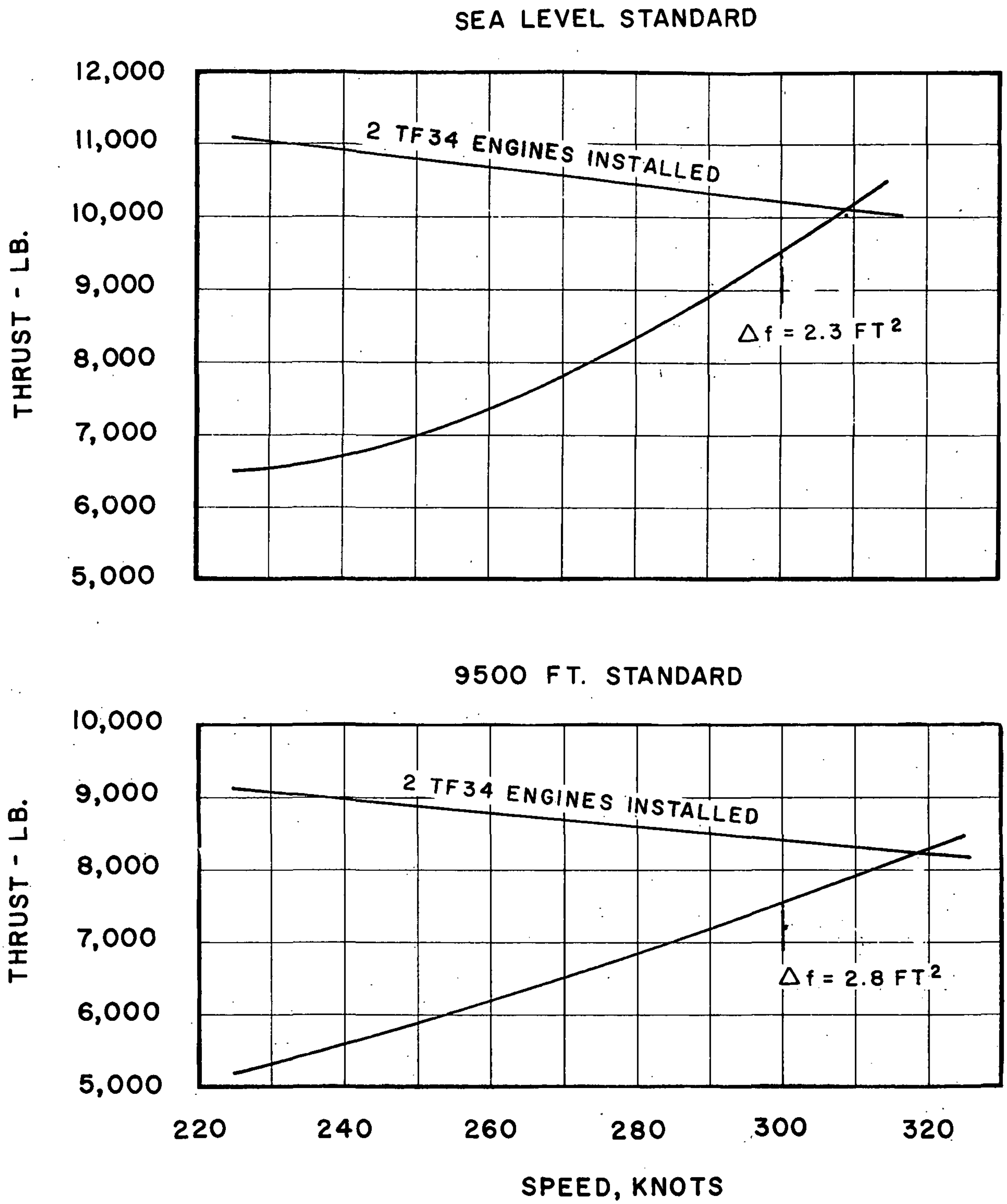
The remaining propulsive force is needed to overcome wing induced drag, that drag associated with the generation of lift. This is evaluated using lift and drag curves developed for the RSRA wing. The curves are based on section characteristics of Reference 7 and were evaluated using the procedures of Reference 8. The procedure is to calculate the lift coefficient, find the required angle of attack from the curve, and read the drag coefficient. Reduce the drag coefficient by .008 since parasite drag of the wing is included in the equivalent flat plate area. Thus:

$$\text{Induced drag} = q \times (S_{wing}) \times (C_D - .008)$$

Thrust available for the TF34-GE-100 turbofans has been obtained from the manufacturer's proposed specification and corrected for additional installation losses.

Aircraft capability exceeds the 300 kt requirement at both conditions. Thus there is the capability to test higher drag rotor systems. The Δf contingency is as follows:

SLS	$\Delta f = 2.3\ ft^2$
9500' Std	$\Delta f = 2.8\ ft^2$



RSRA HIGH SPEED THRUST SMALL WING INSTALLED
THRUST AVAILABLE & THRUST REQUIRED VS. SPEED

FIGURE 62

Mission Analysis

The RSRA mission analysis set up in the compound design model computer program includes fuel flows increased by five percent above the manufacturers' engine performance data and all elements required in the Statement of Work. The mission breakdown for the aircraft is shown as Figure 63 and has all the elements of the Statement of Work included in the fuel calculation. Twenty minutes of fuel at the airspeed for maximum range is the most critical reserve requirement.

TOGW= 26392.0 LBS., ROTOR RADIUS= 31.00 FT., PARASITE DRAG= 23.6 SQ.FT.

TYPE OF ENGINES- NUMBER 9 (2.)

MODE	GR.WT (LBS)	TEMP (DEG.F)	ALT (FT)	OPTN (ZR/FPM)	SPEED (KTS)	VSTALL (KTS)	DIST (N.MI)	TIME (MIN)	FL.AR. (SQ.FT)	SHP	FUEL (LBS)
WU/TO	26392.	59.	0.	--	--	--	--	2.0	--	10190.6	188.1
HOVER	26204.	59.	0.	1000.000	--	--	--	2.0	--	4464.0	106.9
CRUISE	26097.	59.	0.	.00	250.0	*****	8.3	2.0	23.60	8651.1	165.5
DASH	25931.	59.	0.	.00	300.0	*****	75.0	15.0	23.60	12518.4	1781.3
CRUISE	24150.	59.	0.	.00	250.0	*****	8.3	2.0	23.60	8480.7	163.1
HOVER	23987.	59.	0.	1000.000	--	--	--	2.0	--	3853.3	99.0
RESERVE-CRUISE	23888.	59.	0.	.00	146.0	*****	48.7	20.0	23.60	2310.2	809.7

TOTAL MISSION FUEL IS 3313. LBS

TOTAL MISSION TIME IS 25.0 MINS

MISSION BREAKDOWN

FIGURE 63

One Engine Inoperative

Capability of the RSRA in helicopter flight simlation with one T58-GE-16 engine and both TF-34T's inoperative is shown in Figure 64 . At design gross weight with the large wing and turbofans installed, the aircraft minimum speed is 54 knots.

The three dashed lines on Figure 64 represent the aircraft in helicopter flight with wings and turbofans removed. At the operational weight empty of 16074 pounds, the capability to maintain level flight exists from 14 knots to 146 knots. With hover mission fuel, the gross weight is 18276 and the speed range is 25 knots to 141 knots. With the addition of 2000 pounds of payload, the gross weight becomes 20276 pounds, and the speed range is 34 knots to 137 knots.

Power required was calculated using the Sikorsky non-dimensional rotorcraft performance program. Previous work has shown that this approach shows excellent correlation with test data of Sikorsky helicopters in low speed flight. T58-GE-16 power available was obtained from the manufacturer's specifications.

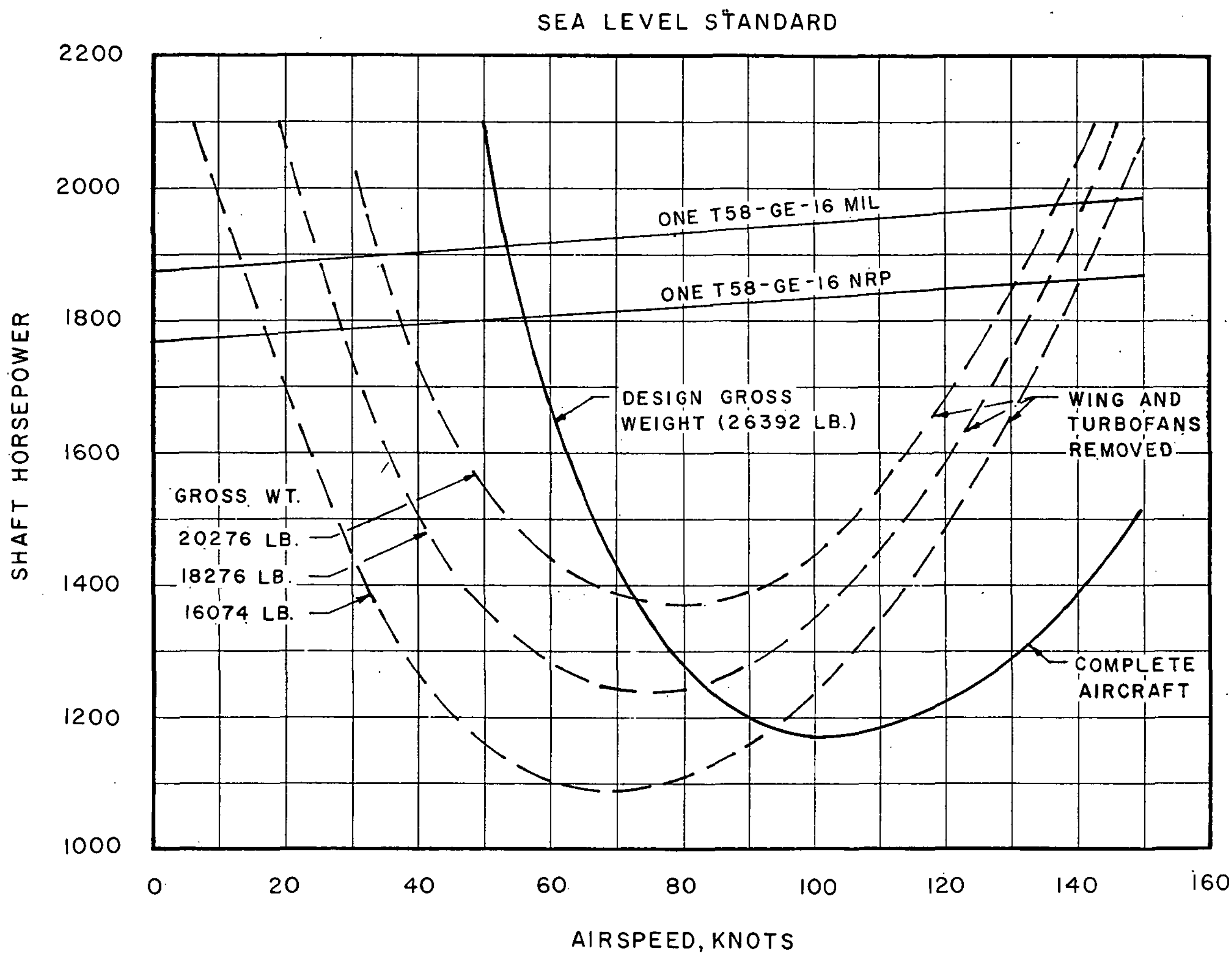


FIGURE 64 **AIRCRAFT PERFORMANCE - ONE ENGINE INOPERATIVE SHAFTHORSEPOWER AVAILABLE & REQUIRED VS. AIRSPEED SEA LEVEL STANDARD**

AIRCRAFT STABILITY AND CONTROL

The stability and control characteristics of the RSRA have been examined with the basic S-67 rotor and control system. The static stability design criteria used for the vertical and horizontal tail is first discussed. This is followed with the trim and dynamic stability characteristics of the RSRA.

Static Stability Criteria

The vertical tail size requirement for the aircraft was based on maintainability at least neutral static directional stability. The analysis was done about the aft cg location. The lift properties of the vertical fin were determined analytically and an estimated correction accounting for the presence of the fan was included. The yawing moment derivative with sideslip for the fuselage was calculated; and this quantity was balanced by the vertical tail. Sidewash and dynamic pressure losses at the vertical tail were included in the analysis. The area needed to obtain neutral stability was found to be 83% of the actual area designated for the aircraft.

Typically, helicopter vertical tail size is determined based on neutral stability. Some positive stability margin is desired, and this is usually provided by the tail rotor. The RSRA design employs a fan rather than a tail rotor, however the positive margin should still be available. Presently, little empirical data exists describing the effect of the fan thrust on the lift curve slope of the fin. It is known that the slope decreases as the fan thrust is reduced. Therefore, the selected vertical tail size would be considered adequate without the additional surface area gained by closing the fan duct with a shutter mechanism. The necessity of either increased area or covering the fan openings to provide for neutral stability in event of a failure of the fan should be further investigated when more data becomes available on the effect of fan thrust on fin lift. A fan shutter mechanism is included in the aircraft design.

The design condition for sizing the horizontal tail was the ability to land the RSRA at design gross weight in the pure conventional aircraft mode. For this condition, it was assumed that the main rotor produced only drag. The most critical configuration selected was the forward cg with full flap deflection. Two speeds were studied; 120 knots and 95 knots. The latter is a minimum speed corresponding to the maximum obtainable lift coefficient of the flapped wing. The resulting horizontal tail size requirement as a function of wing incidence is shown as Figure 65 . Plots are shown for the tail operating at its maximum lift capability, and at lower lift coefficients which allow for control and stall margins. Horizontal tail incidence limits are +20 degrees to -25 degrees.

RSRA HORIZONTAL TAIL SIZING BASED ON LANDINGS WITH FULL FLAPS IN PURE FIXED WING MODE (INCL. DRAG FROM ROTOR)

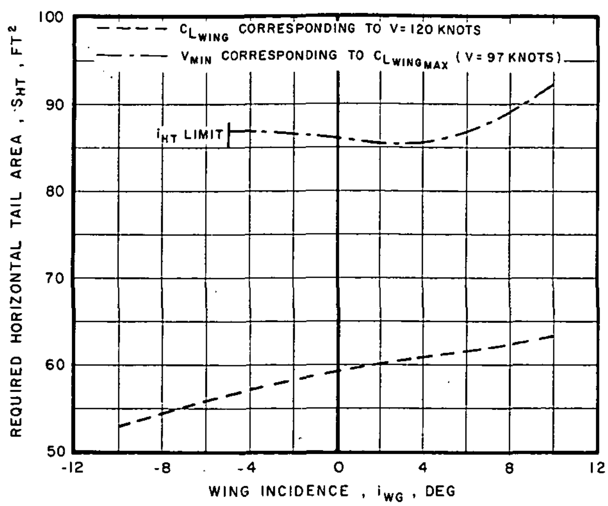


FIGURE 65

The data presented were gathered by determining the lift at the horizontal tail needed to counter the pitching moment produced by the wing, fuselage, and rotor. Thus, these data represent trim criteria. The RSRA exhibits positive pitching moment with angle of attack stability for the landing cases studied. Neutral stability about the aft cg for the unflapped wing and thrusting main rotor condition was also investigated and found to yield a horizontal tail size requirement of 43.5 ft^2. Thus, the landing condition is the most critical for tail design. Dynamic pressures losses, fuselage downwash, and induced flow at the tail due to the bound and shed vortices of the wing were all considered in the horizontal tail analysis.

Fan-in Fin Capability

An analysis was conducted to check the capability of the fan-in-fin on the RSRA aircraft in hover. The analysis consisted of comparing the thrust and power requirements of the fan in the RSRA and the S-67. Since the RSRA critical design point is sea level, 95° and the S-67 critical point is 4000' 95°, the higher gross weight of the RSRA is somewhat neutralized by the lower design density altitude. The study compared the two configurations against the one inch input requirements of MIL-8501A; the RSRA C_T/σ operating point is only 5% more than the S-67. This is well within the capability of the fan. The power requirement increase was about 35%, but is still well within the capability of the fan and the fan gearbox. With the T58-GE-16 engines installed, enough power is available to the RSRA fan.

Since the RSRA is a rotor test vehicle, the side flight requirements of MIL-H-8501A need not necessarily apply to this aircraft. Under this condition, the intent of MIL-H-8501A will be met.

The trim and dynamic stability characteristics of the RSRA were generated using Sikorsky's Generalized Helicopter Simulation Program. The General Helicopter Simulation Program is programmed on a hybrid computer and includes the following:

1. Six rigid body fuselage degrees of freedom of motion with no simplifying assumptions.

2. Nonlinear fuselage, wing, vertical tail and horizontal tail aerodynamic data.

3. Nonlinear rotor force and moment equations of motion for each blade with no angle of attack or advance ratio restrictions

4. Nonlinear rotor blade airfoil section aerodynamic data, including stall and compressibility effects.

5. Full representation of control system.

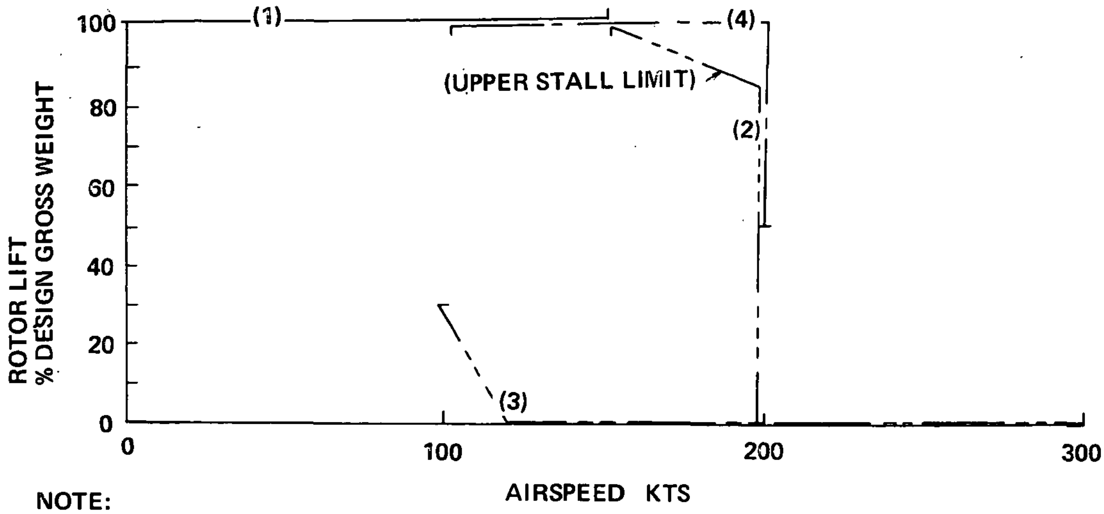
Trim

Unlike a helicopter, for which a single rotor trim control setting is unique for a given flight condition (i.e., speed, altitude, gross weight, and center of gravity location) the compound helicopter can be flown at a wide variety of rotor and fixed wing control settings, for the same flight condition. This is because a different wing, rotor lift and rotor drag combination can be obtained by changing body/wing angle of attack through elevator control, by changing the wing angle of attack through wing incidence control, or the wing lift coefficient through wing flap control, and by adding auxiliary propulsion to alter the rotor propulsive force.

To simplify the task of examining the aircraft's handling qualities, some constraints on aircraft attitude were imposed, which it is believed closely reflect the manner in which the aircraft will be flown during its design mission. In the conventional helicopter flight mode, the wing incidence was adjusted to produce no lift and the elevator was set to maintain a desired degree of rotor flapping. In the compound mode, the aircraft pitch attitude was held level and auxiliary propulsion and wing lift set as required to examine boundary conditions. Level, autorotative flight was examined at full low collective, and a fixed degree of rotor flapping.

The boundaries of various modes of trim flight are shown in Figure 66.

In the region from 100 to 200 knots, the speed range specified for helicopter flight, the control positions for trimmed flight were first determined for full rotor loading (100% of gross weight and 100% of propulsive force) until the upper stall limit of the rotor (S67 boundary) was reached. For this mode of operation, the flight attitude of the aircraft was varied to maintain zero roll angle, and the only constraint on the rotor system was to maintain zero (or near zero) rolling moment, and a specified range of pitching moment. The determination of control positions for operation along the rotor



NOTE:

(1) ALL LIFT AND PROPULSIVE FORCE FROM THE ROTOR LIMITED BY THE ROTOR UPPER STALL LIMIT
(2) ROTOR LIFT AND PROPULSIVE FORCE AUGMENTED BY WING LIFT AND FAN-JET PROPULSION
(3) PARTIAL ROTOR LIFT FROM 100 TO 120 KNOTS, FULL WING LIFT FROM 120 TO 300 KNOTS: ALL PROPULSIVE FORCE FROM TURBOFAN ENGINES
(4) LEVEL, AUTOROTATIVE FLIGHT WITH PROPULSION FROM TURBOFAN ENGINES

RSRA "HELICOPTER SIMULATION" BOUNDARIES

BASELINE S-67 ROTOR INSTALLED

FIGURE 66

upper stall limit required that the rotor lift and propulsive forces be supplemented by wing lift and fan propulsion so that the rotor can be held at the upper stall limit as the aircraft speed is increased. This mode of operation required the maximum rotor control travel. Trimmed flight was analyzed and control positions determined at a zero fuselage pitch and roll attitude, again with a zero (or near zero) rotor rolling moment, and a specified range of rotor pitching moment. The fuselage attitude was held level; however this can be varied by elevator control within the design flapping limits of the rotor system.

Other operating conditions studied were autorotation and minimum rotor lift. In autorotation, rotor lift was maintained at 100% of aircraft gross weight in level flight. At minimum rotor lift, the rotor was operated at zero lift except at low speed where the wing was incapable of supporting the total aircraft weight.

In these modes of operation, the S-67 rotor trim was maintained within the design flapping and control limits. The fuselage pitch attitude was held at zero degrees for flight at maximum wing lift, and in the range between zero and 5 degrees nose-up for autorotative flight.

This trim study showed that the S-61 control system used for the RSRA provides adequate collective and lateral control range to operate along the flight boundaries previously discussed. The longitudinal cyclic control position varies most widely to trim the aircraft for the desired range of operating airspeeds and rotor loading conditions. For this reason, the longitudinal cyclic stick positions are only shown (Figures 67 to 68) to focus on the capability of this rotor and rotor control system to fly at all desired test airspeeds.

The longitudinal control positions for trimmed flight at 100% rotor lift and propulsive force are shown on Figure 67. At 26392 lbs gross weight, the upper stall limit is reached at about 150 knots. Higher speeds at the upper stall limit are attained at reduced rotor thrust by use of auxiliary propulsion and wing lift. The control required for trim at these conditions is shown in Figure 68. For this flight boundary, adequate control margin is available, but a change in fuselage pitch attitude or rotor flapping will be required for trim to maintain longitudinal control margins. The data shown on Figure 68 is for a zero fuselage pitch attitude.

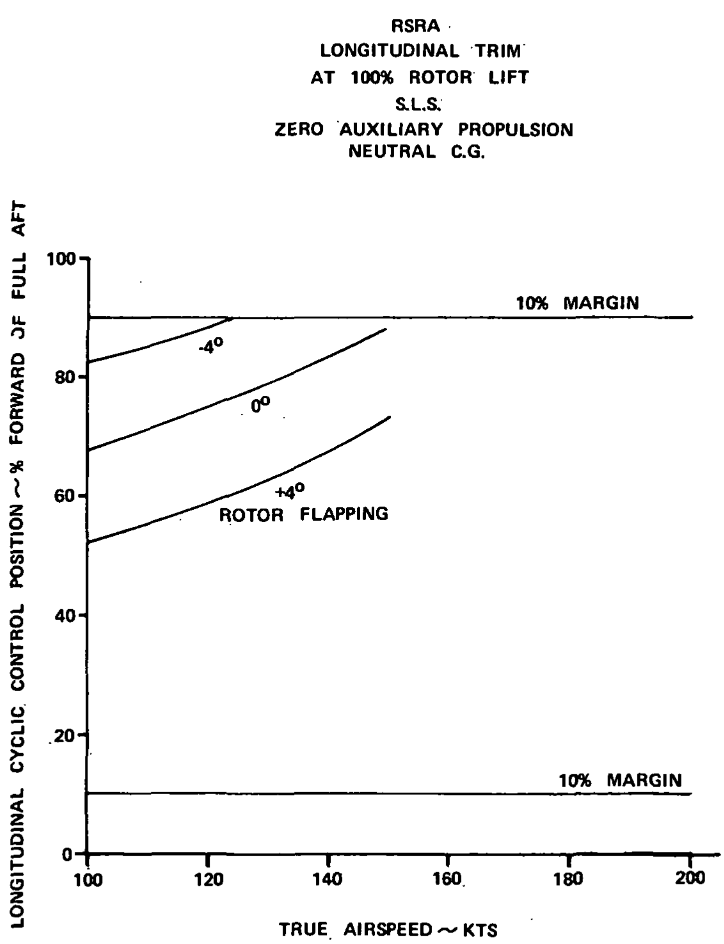


FIGURE 67

The effect of varying elevator deflection is shown on the figures as rotor flapping (a_{1s}) limits. The design limit of the S-61 rotor is ±4°. Ordinarily, the degree of rotor flapping is associated with a center of gravity displacement in a helicopter having a fixed horizontal stabilizer. In the RSRA, however, the elevator is used as needed to trim the fuselage attitude, so the allowable range of center of gravity travel is no longer a function of rotor flapping.

The rotor and aircraft trim requirements were also examined at zero rotor lift between 120 and 300 knots and the results are shown in Figure 69. The aircraft attitude in this range of flight speeds has been held to zero pitch and zero roll angles. Adequate control margins are available at both speed extremes

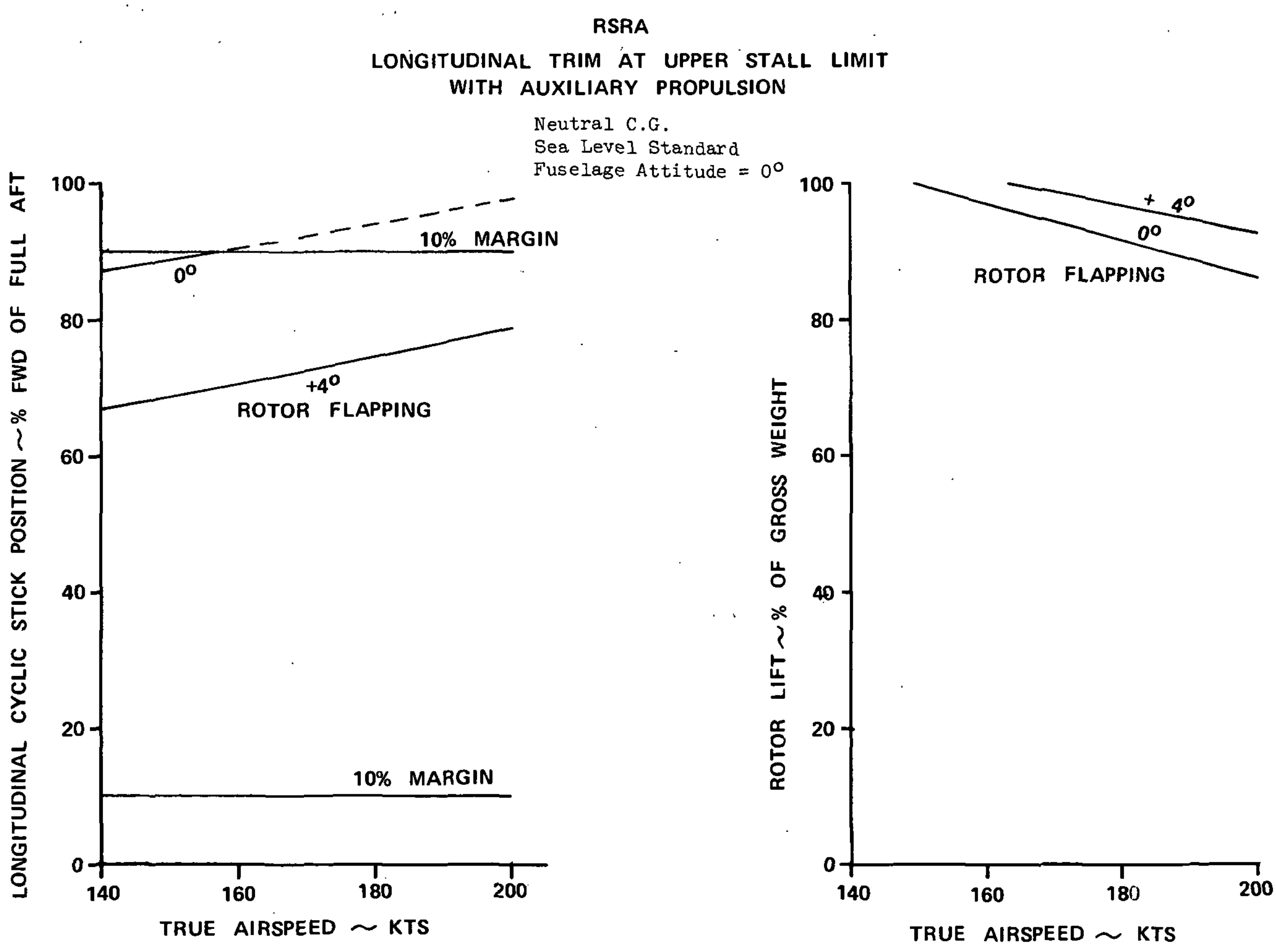


FIGURE 68

RSRA
LONGITUDINAL TRIM
AT ZERO ROTOR LIFT
S.L.S.
NEUTRAL CG

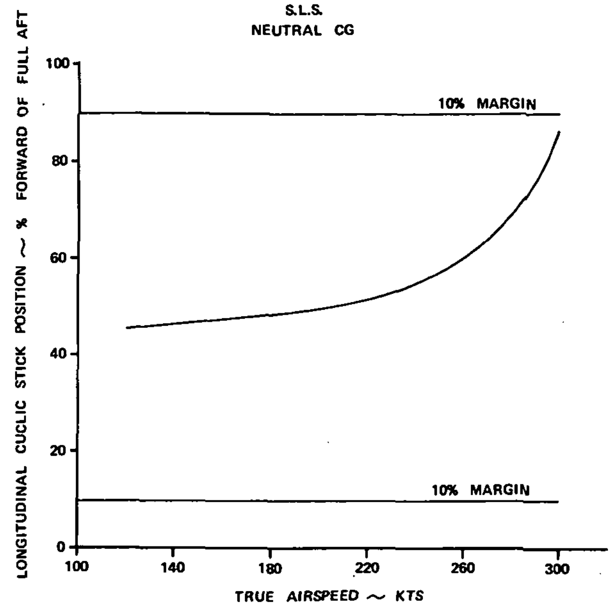


FIGURE 69

RSRA
LONGITUDINAL TRIM
LEVEL FLIGHT AUTOROTATION
USING AUXILIARY PORPULSION
100% ROTOR LIFT, +4° FLAPPING
NEUTRAL CG

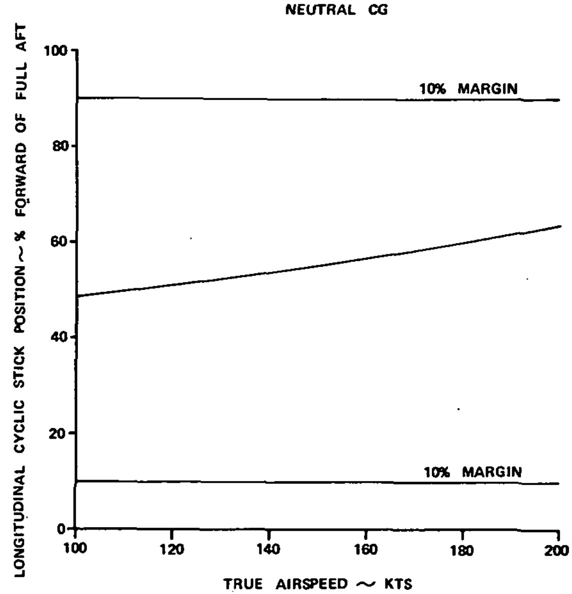


FIGURE 70

and there are no longitudinal cyclic stick control reversals throughout this speed range.

The RSRA also has the capability for sustained, level, autorotative flight. Auxiliary propulsion is used to sustain the selected speed, and the longitudinal cyclic stick is used to trim for level flight. The trim positions for level, autorotative flight between 100 and 200 knots are shown on Figure 70 These were determined for the rotor at 100% lift and at the full low collective control position. In the speed range examined, the fuselage pitch attitude varied from about 4.5° to 6.5°; the main rotor flapping was limited to $+4^{\circ}$.

The longitudinal cyclic control has no reversals in level autorotative flights, and control margins satisfy the requirements of the MIL-H-8501A specification.

Longitudinal trim was also examined for autorotative descent in the helicopter mode, between 60 and 100 knots. The longitudinal trim position and rates of descent for full low collective blade setting are shown on Figure 71 Zero wing lift is maintained through independent wing incidence control.

From the examination of the longitudinal cyclic stick trim positions shown in Figures 67 through 71 it is concluded that:

1. The flight envelope described in Figure 66 can be flown with adequate longitudinal control margin in accordance with the requirements of MIL-H-8501A.

2. The main rotor longitudinal flapping can be held within design limits by controlling the fuselage pitch attitude with the appropriate elevator setting, throughout the flight envelope described in Figure 66.

3. The RSRA possesses adequate longitudinal trim margin to execute normal autorotative descents between 60 and 100 knots.

Dynamic Stability

The dynamic response of the RSRA was examined at flight speeds and rotor thrust conditions which represent the operating boundaries of the RSRA with the S-67 rotor system. Time histories of the aircraft response to longitudinal and lateral cyclic stick pulses were generated with the Sikorsky Generalized Helicopter Simulation Program.

Two independent electro-mechanical systems augment the dynamic stability of the RSRA. They are a Stability Augmentation System and the computer. During non-test conditions the Stability Augmentation System is used. When the RSRA is used as a test bed for rotor system testing the onboard computer holds the selected flight condition and takes over the function of the SAS which is placed on standby.

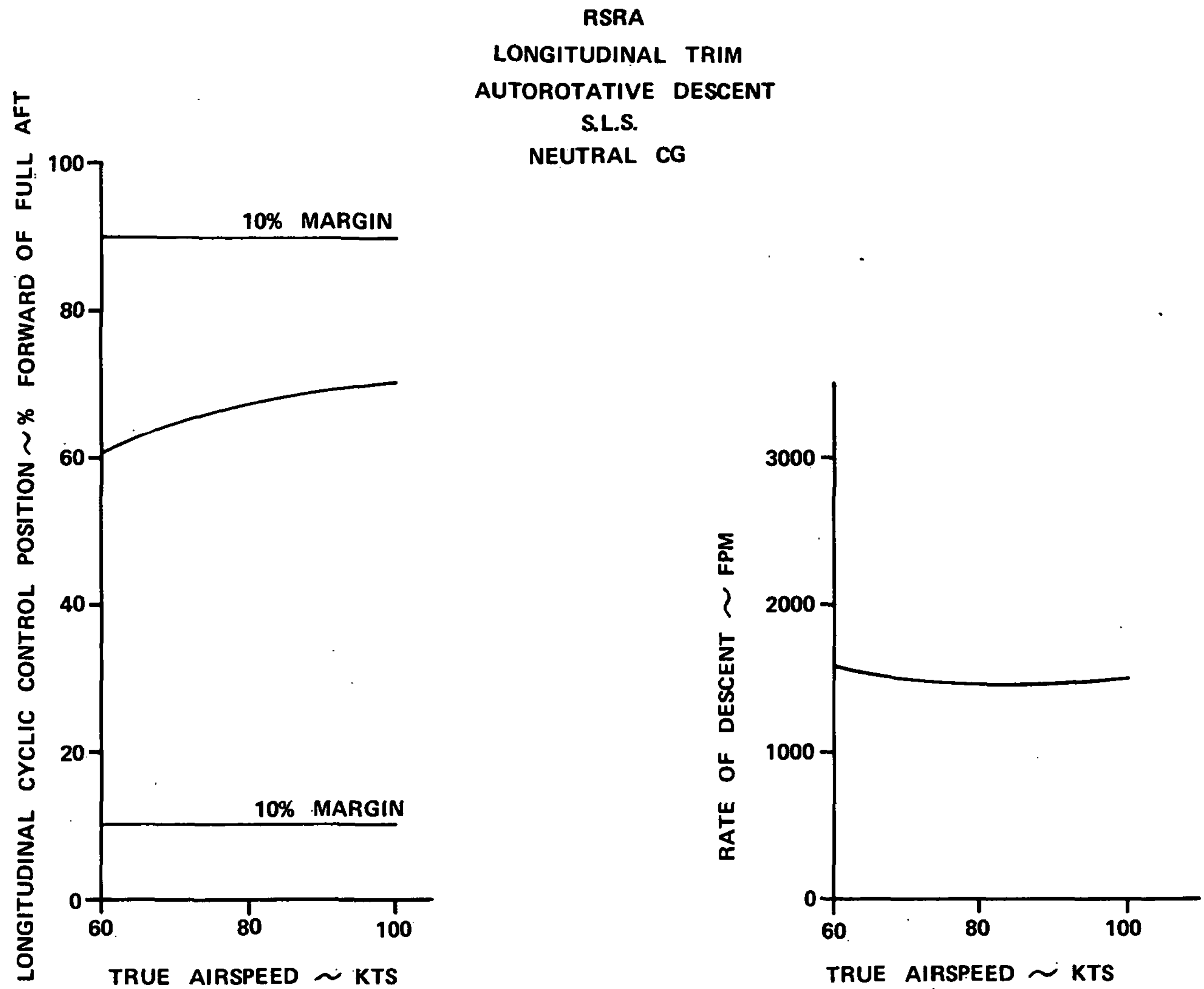


FIGURE 71

During the normal flight mode, the Stability Augmentation System (SAS) will provide the basic aircraft stability through the limited authority auxiliary servos, which will move both the rotor and conventional fixed wing control surfaces. In this flight mode, the RSRA will have flight characteristics similar to the S-67 and is designed to meet MIL-H-8501A.

The response of the RSRA to an aft longitudinal pulse input is shown (Figures 72 through 75) for a neutral center of gravity position while operating in the test mode. During this mode of operation, the rotor is maintained at a preselected test condition (i.e., given lift and angle of attack) via the onboard computer and the control system.

Figure 72 shows that the aircraft is stable with the rotor carrying 100% of the lift and supplying all the required forward propulsive force for speeds of 40, 100, and 150 knots. The control input is a one inch aft stick displacement, held for 1 second and returned. The shape of the B_{1s} curve in the figures represents the shape of the control input at the rotor head which results from a combination of the stick input and inputs from the onboard computer and control system. The RSRA meets the MIL-H-8501A specification requirement that the pitch acceleration be in the direction of the control input within 0.2 seconds of the stick movement.

The ability to meet the same requirement while operating at the upper rotor stall limit at 200 knots is shown in Figure 73. The rotor lift is 22,400 pounds, with the remainder of the gross weight supported by the wing. Auxiliary propulsion from the fans supplements rotor propulsive force to maintain level flight at this speed and flight condition.

The examination of the RSRA ability to satisfy the MIL-H-8501A requirement was also carried to the condition of zero rotor lift. (At 120 knots, full wing flap is required to support the weight of the aircraft.) Aircraft response to longitudinal stick movement at zero rotor loading was also examined at 200 and 300 knots. The results at all three speeds of this flight condition are shown in Figure 74 to satisfy MIL-H-8501A.

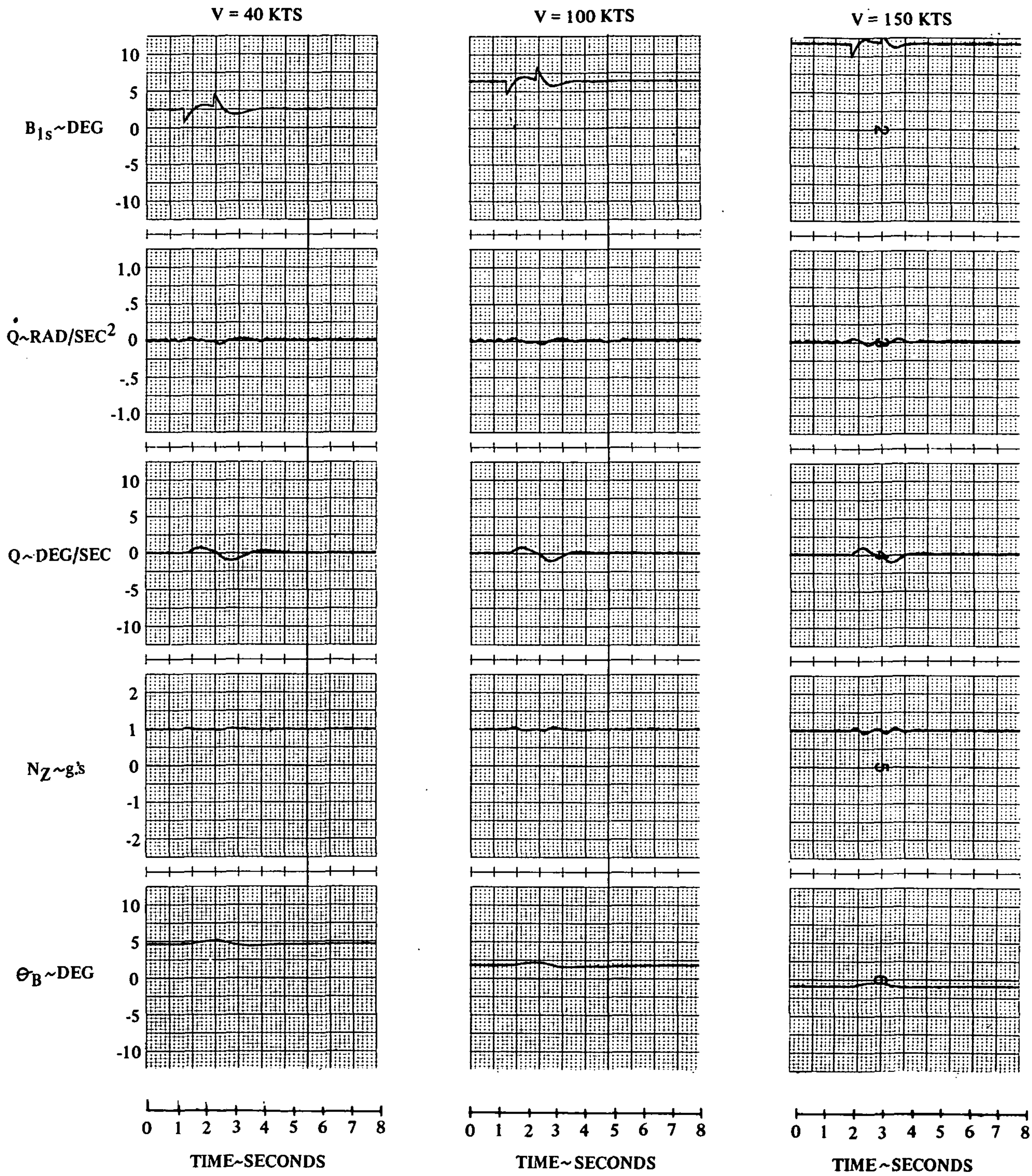
The aircraft reaction to a cyclic pulse was also examined with the aircraft in level, autorotative flight. The results, shown in Figure 75 at 100 and 200 knots, satisfy the response requirements of MIL-H-8501A specification.

The lateral coupling to longitudinal displacement was negligible in all cases investigated.

Aircraft response to a one inch lateral cyclic displacement, held for 1 second and returned, at 200 knots airspeed, is shown in Figure 76. At this flight condition, the rotor is operating at the upper stall limit, with 22,400 pounds supported by the rotor. Auxiliary propulsion from the fans supplements the rotor propulsive force. The characteristic reaction to the control input and the return to a trimmed condition is typical of the response to a lateral pulse over the whole range of flight conditions studied.

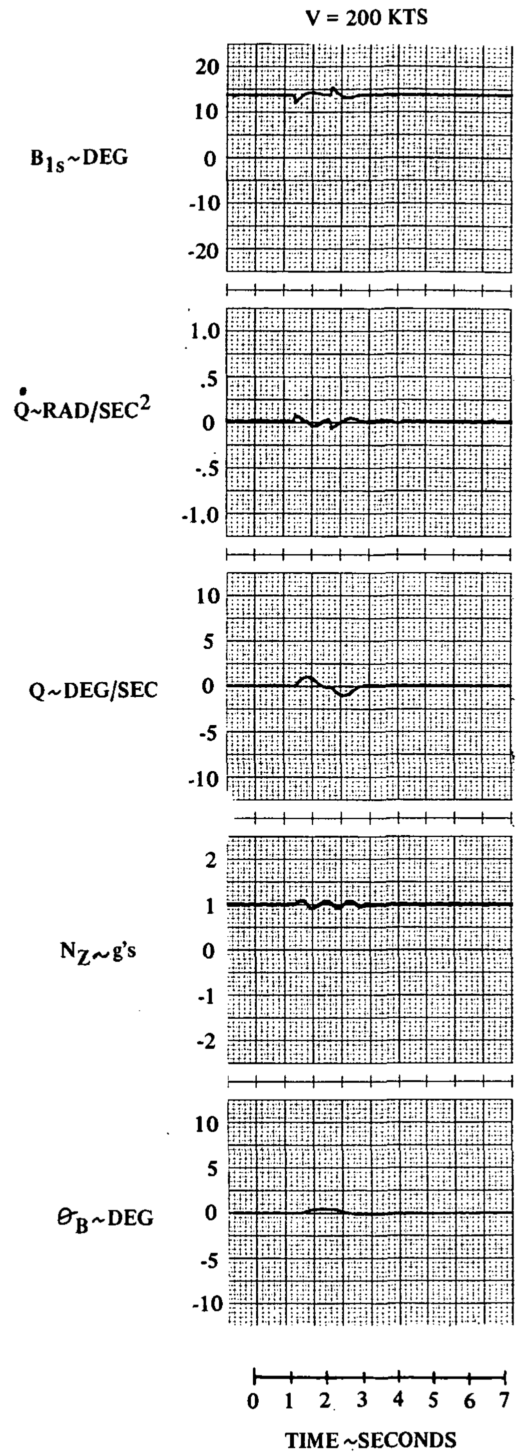
The yaw response of the RSRA has been evaluated in hover and found to meet the intent of MIL-H-8501A. Response to yaw inputs in forward flight could not be obtained, because the thrust control characteristics of the yaw control fan in forward flight are not presently available. These are being developed under Army Contract DAAJ02-72-C-0050. This aspect of the RSRA handling qualities will have to be investigated during preliminary design, after the development of the yaw fan.

It is concluded from this preliminary dynamic stability study that the RSRA when operating in the test mode will provide a stable platform and is well suited as a rotor test vehicle.



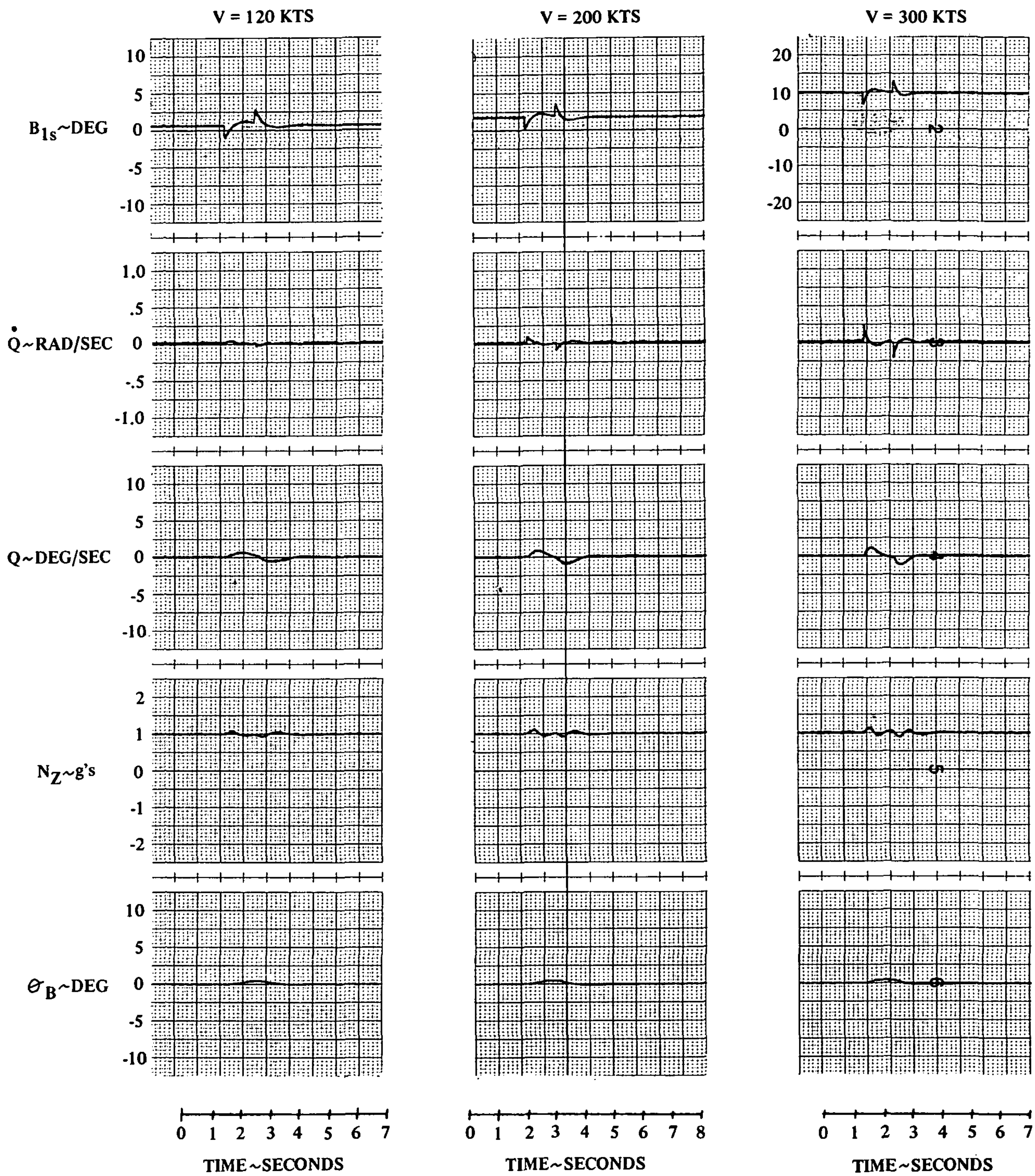
RSRA PITCH RESPONSE TO A ONE INCH AFT CYCLIC PULSE AT
100% ROTOR LOADING, NEUTRAL C.G. (TEST MODE)

FIGURE 72



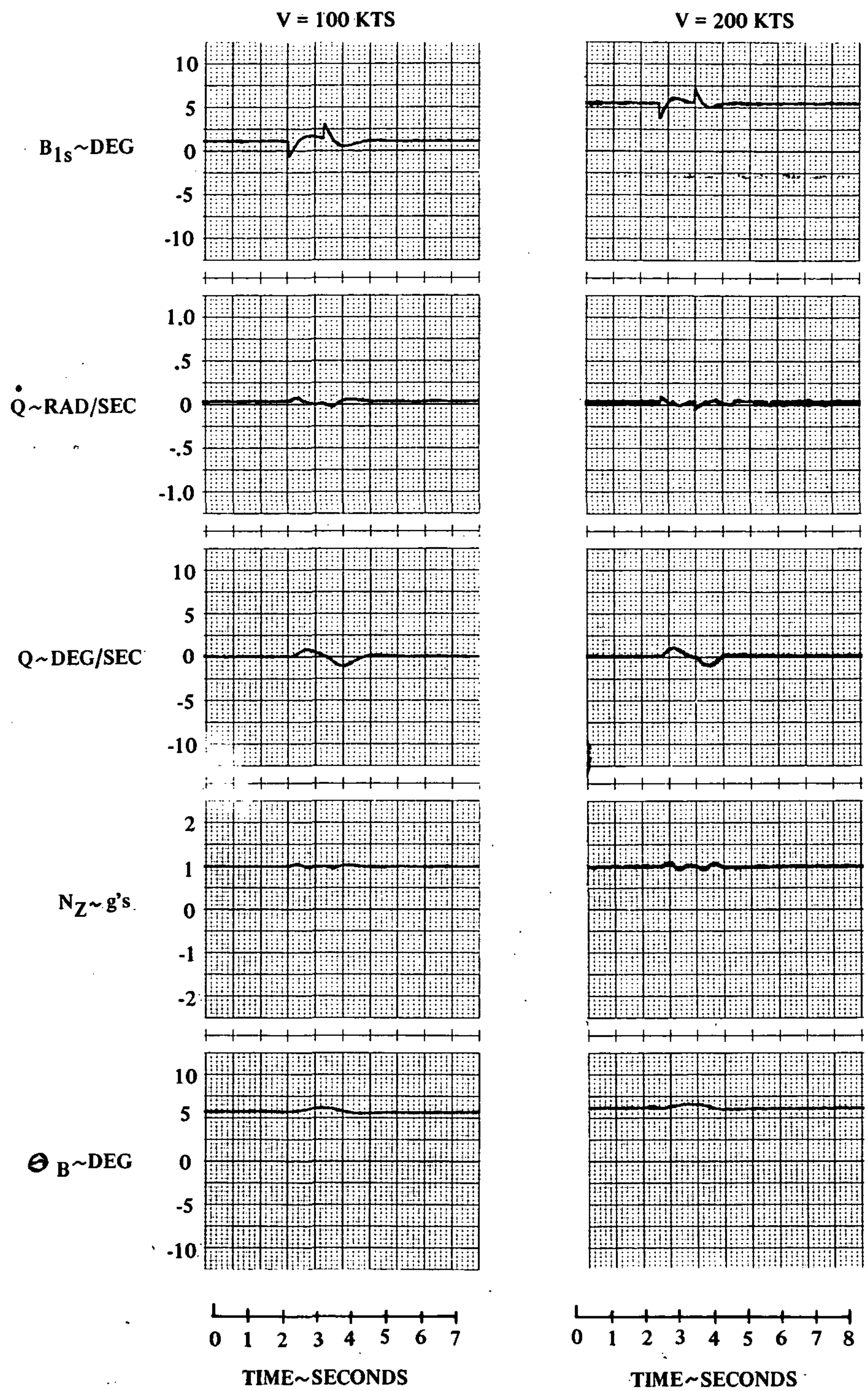
RSRA PITCH RESPONSE TO A ONE INCH AFT CYCLIC PULSE AT PARTIAL ROTOR LOADING, NEUTRAL C.G. (TEST MODE)

FIGURE 73



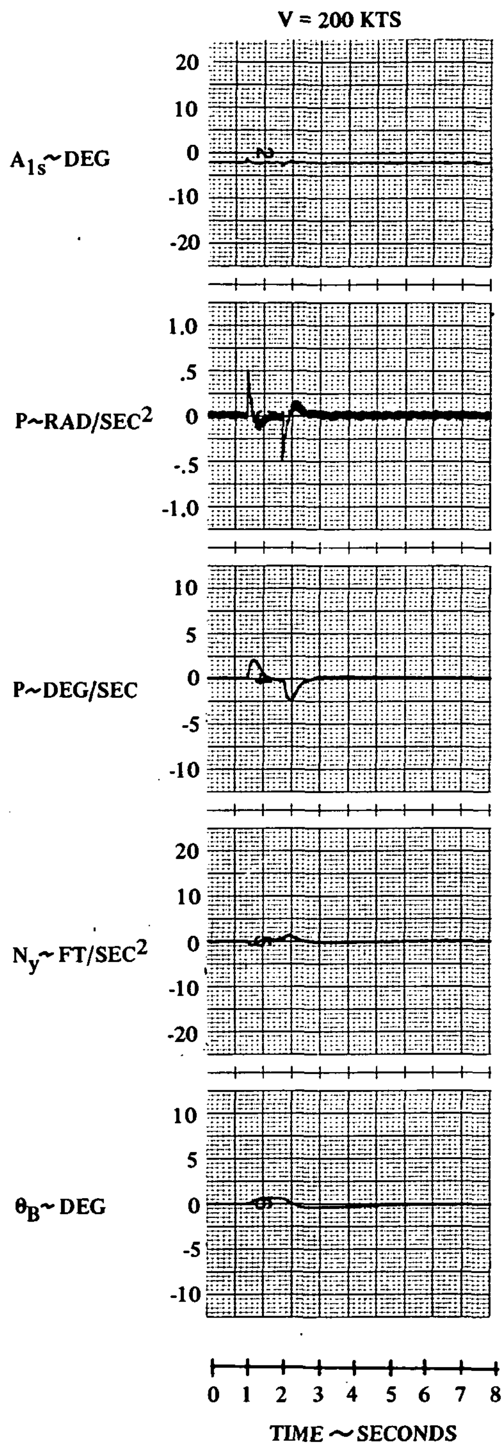
RSRA PITCH RESPONSE TO A ONE INCH AFT CYCLIC PULSE AT MAXIMUM WING LOADING, NEUTRAL C.G. (TEST MODE)

FIGURE 74



RSRA PITCH RESPONSE TO A ONE INCH AFT CYCLIC PULSE IN
LEVEL AUTOROTATIVE FLIGHT; NEUTRAL C.G. (TEST MODE)

FIGURE 75



RSRA ROLL RESPONSE TO A ONE INCH LATERAL CYCLIC
PULSE AT PARTIAL ROTOR LOADING, NEUTRAL C.G. (TEST MODE)

FIGURE 76

The primary function of RSRA is to test a wide range of advanced helicopter and compound rotor systems. It must accept these rotors with minimum modification to the basic aircraft.

The rotor systems receiving primary emphasis during the Predesign Study were the baseline compound rotor and the variable geometry rotor. In addition to these, other rotor systems, representing a broad range of rotor types which might be tested on RSRA, were also considered in the design of the basic vehicle. These included:

(1) Variable diameter rotor
(2) Variable twist rotor
(3) Coaxial rotor
(4) Jet flap rotor
(5) Slowed rotor

A summary of the aircraft modifications required to accommodate these rotors are listed in Figure 77. Topics considered are drive system modifications, engine modifications, control system modifications, and whether an RPM variation of greater than 30 percent is required.

ROTOR CONFIGURATION	NEW MAIN GEARBOX	NEW ENGINE INSTALLATION	CONTROL SYSTEM	RPM VARIATION GREATER THAN 30% *
RIGID COUNTERROTATING COAXIAL ROTOR	YES	NO	MAJOR	NO
VARIABLE DIAMETER ROTOR	NO	NO	MINOR	NO
JET FLAP ROTOR	YES	YES	MAJOR	NO
VARIABLE TWIST ROTOR	NO	NO	MINOR	NO
SLOWED ROTOR	NO	NO	MINOR	YES

* WILL REQUIRE ACTIVE VIBRATION SUPPRESSION PLUS

MODIFICATIONS FOR ALTERNATE ROTORS

FIGURE 77

All of the single rotor shaft driven concepts use the same main gearbox and the engine installation. The rigid counterrotating coaxial rotor requires a new gearbox, driven by the same engine installation. The jet flap rotor requires gas generators instead of shaft power engines and a main rotor support structure replacing the conventional mechanical gearbox. This rotor support structure is combined with a new RSRA accessory gearbox with an output shaft for the fan-in-fin drive, used for yaw control.

All of the concepts require some modification to the rotor control system. This involves modifying the rotor mixing unit and some modifications between the mixing unit and the rotor itself. The rigid coaxial requires modifications to the control system to provide control for two rotors. The variable diameter rotor, the variable twist rotor, and the slowed rotor use the baseline system with minor modifications. With the variable diameter rotor a separate control is provided for the rotor diameter. For the variable twist rotor, separate control of twist is obtained by a second control assembly for outboard pitch control. A completely different control system beyond the mixing unit is required for the jet flap rotor.

The final item on the chart considers whether an RPM variation of over 30 percent is required in the operation of the rotors being considered. If such an RPM variation is required, the active vibration suppression system will be needed to avoid rotor/airframe dynamic resonance.

The Variable Geometry Rotor

The design for the variable geometry rotor is shown on Figure 78 . Two three-bladed rotors are located on the same rotor shaft. These can be displaced vertically at three different positions to test the effect of vertical displacement on rotor performance. The upper rotor can also be indexed with respect to the lower to study the effects of non-uniform blade azimuth position. This indexing is done by means of the spline between the hub and the rotor shaft. The rotating swashplate is designed such that the blade control rods can be repositioned with the blades.

The design is based on making maximum use of existing S-61 components, tooling and inspection gages. The only new parts required are three-bladed hub plates, upper hub shaft and spacers, plus the rotating swashplate and pushrods. The shaft splines, threads, bores, tapers, etc. for the new parts are the same as on S-61 standard parts. Grease lubrication instead of oil will be used throughout the hub assemblies to permit closer vertical spacing. Close azimuth spacing is made possible with a special damper which positions all the blades against the lead stops for starting and stopping.

The complete assembly will consist of two hub assemblies and three sets of pushrods and spacers to accommodate all three upper hub positions. A set of six-bladed hub plates is included for coplaner baseline testing.

The variable diameter rotor can duplicate the range of small scale wind tunnel testing accomplished by United Aircraft Research Laboratories under government contracts. In addition, the lower hub can be used separately for three-bladed rotor research.

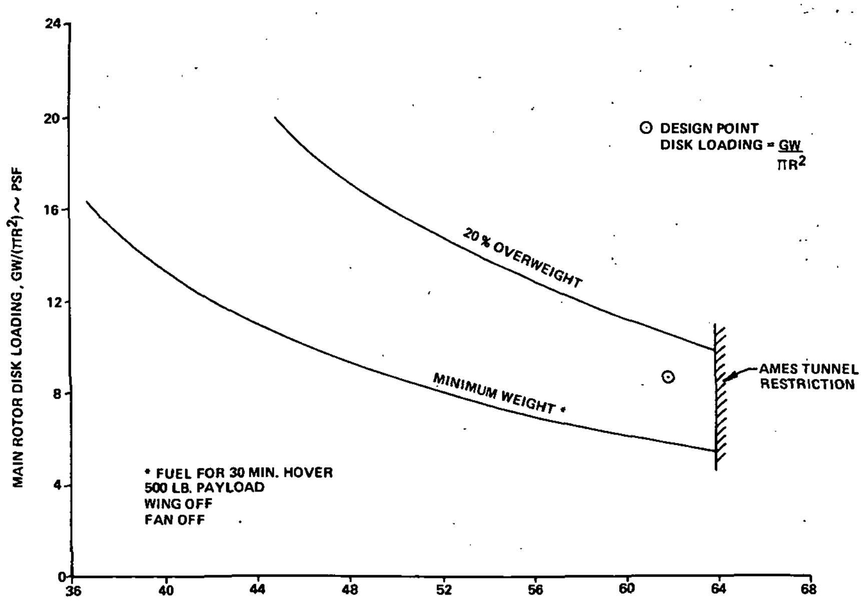
The possibilities of inflight geometry variation were considered. Although it would be feasible to provide such a capability with each case requiring different mechanisms and degrees of complexity, it does not appear that the advantages of inflight variation warrant it's additional cost when compared to a ground adjustable system.

Range of New Rotor Diameters

A study was conducted to establish the range sizes and disk loadings which can be tested on the RSRA. Figure 79 shows the variety of disk loadings which may be tested by varying the RSRA main rotor diameter. The circled point represents the design condition; i.e. 26,392 lb gross weight and a 62-foot diameter rotor.

The upper line in Figure 79 represents a constant gross weight, 20% above the design value, or 31,670 lb.

The lower line is the minimum operating weight. This weight is defined with the wings and fans removed, no payload, and fuel only for the thirty minutes hover mission. With the 62-foot rotor system, this weight is 18,276 lb. At other rotor diameters adjustments are made based on weights information to reflect the changes in rotor and drive system weight and hover fuel requirements.



RSRA CAPABILITY TO TEST A RANGE OF DISC LOADINGS
FIGURE 79

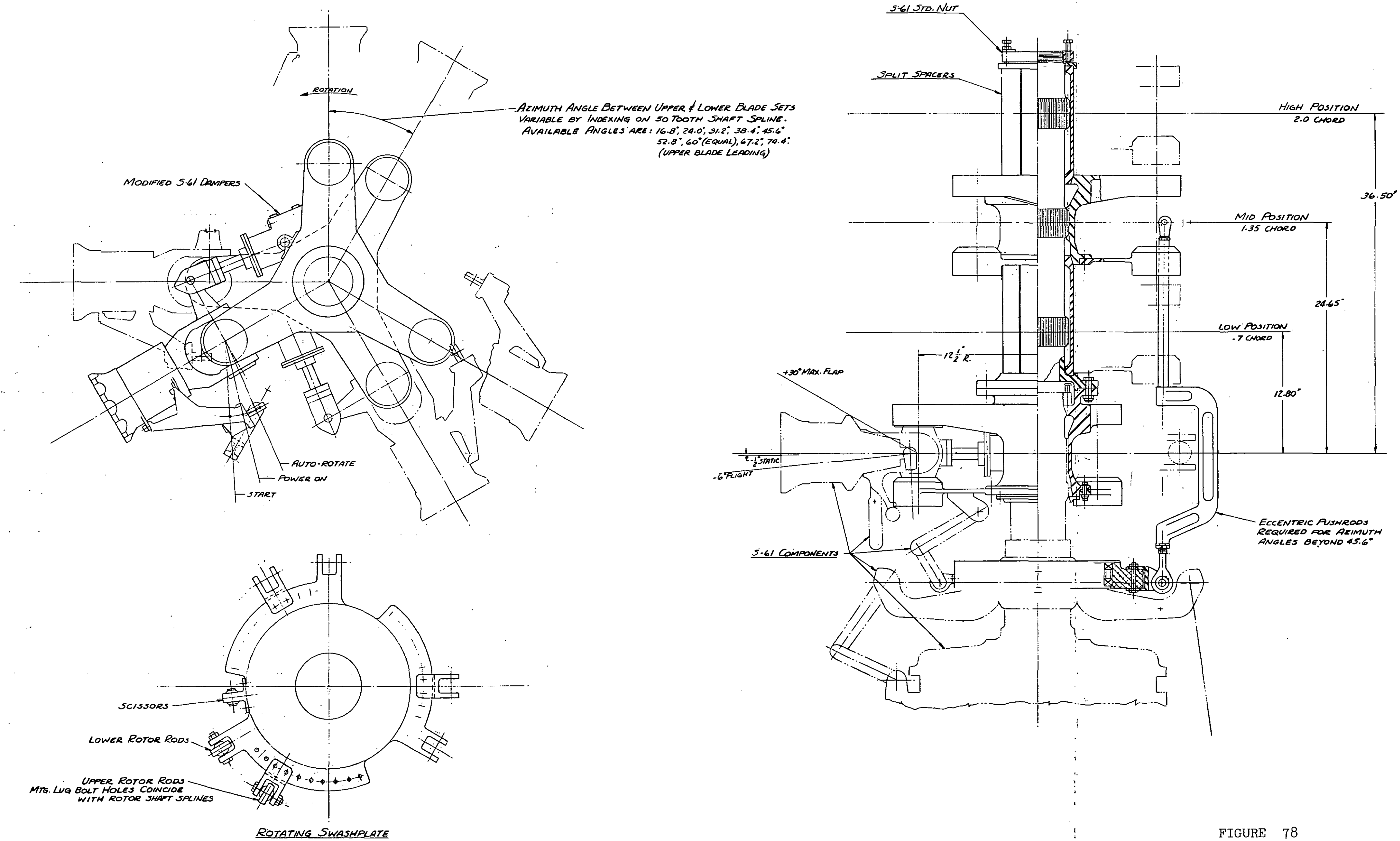

ROTATION
AZIMUTH ANGLE BETWEEN UPPER & LOWER BLADE SETS
VARIABLE BY INDEXING ON 50 TOOTH SHAFT SPLINE.
AVAILABLE ANGLES ARE: 16.8°, 24.0°, 31.2°, 38.4°, 45.6°
52.8°, 60° (EQUAL), 67.2°, 74.4°.
(UPPER BLADE LEADING)
MODIFIED S-61 DAMPERS
AUTO-ROTATE
POWER ON
START
SCISSORS
LOWER ROTOR RODS
UPPER ROTOR RODS
MTG. LUG BOLT HOLES COINCIDE
WITH ROTOR SHAFT SPLINES
ROTATING SWASHPLATE
S-61 STD. NUT
SPLIT SPACERS
HIGH POSITION
2.0 CHORD
36.50"
MID POSITION
1.35 CHORD
24.65"
LOW POSITION
.7 CHORD
12.80"
+30° MAX. FLAP
STATIC
-6° FLIGHT
S-61 COMPONENTS
ECCENTRIC PUSHRODS
REQUIRED FOR AZIMUTH
ANGLES BEYOND 45.6°

FIGURE 78

The Ames tunnel restriction is not a definite boundary, but is Sikorsky's best estimate. A Sikorsky 56 ft. diameter rotor has been successfully tested in the Ames facility. We believe that good results can be obtained up to approximately 64 feet diameter.

Since the blade loading is high on the basic design, hover figure of merit and performance may be improved as needed by increasing solidity (chord or number of blades). Smaller blades areas ($C_T/\sigma > .115$) will probably not be required to test over a normal blade loading range.

A range of disk loading from 5.8 to 10.5 psf can be tested with minimum modification to the aircraft. Figure 80 is a chart of maximum and minimum available disk loadings. Other factors, which have not been included here, must be considered when the rotor system is dramatically altered. Higher disk loadings resulting from reduction in the rotor diameter will increase vertical drag and hence have an adverse effect on performance. Higher power requirements mean stronger, and heavier transmissions. Lower disk loadings employ increased diameter rotors, which will require extension of the tailcone.

	GROSS WEIGHT	ROTOR DIAMETER	DISK LOADING
LOW DISK LOADING			
MINIMUM GW	17700 LB	64.0 FT	5.5 PSF
MAXIMUM GW	31670 LB	64.0 FT	9.85 PSF
HIGH DISK LOADING			
MINIMUM GW	17700 LB	33.6 FT	20 PSF
MAXIMUM GW	31670 LB	44.9 FT	20 PSF

RSRA MAXIMUM & MINIMUM DISC LOADINGS

FIGURE 80

Rotor Response Studies

The feedback control system required to command and maintain the desired rotor parameters was designed using linear control techniques and nonlinear hybrid computer simulations. The RSRA rotor was simulated on the Sikorsky hybrid computer. This simulation is nonlinear and assumes constant RPM, rigid blades and uniform inflow. All other nonlinear terms are included. To simplify the feedback design process, it was decided that a linear approximation of the rotor response would allow an adequate initial design to be established. Frequency response data was gathered at three airspeeds, 100, 200, and 300 knots. Transfer functions were obtained by fitting the rotor frequency response data to first and second order frequency response curves. The resultant transfer functions are given in TableXIVand the pitch moment frequency response data is shown in Figure 81. Note that the responses at the various airspeeds differ only in magnitude. The amplitude and phase (not shown) curves are constant.

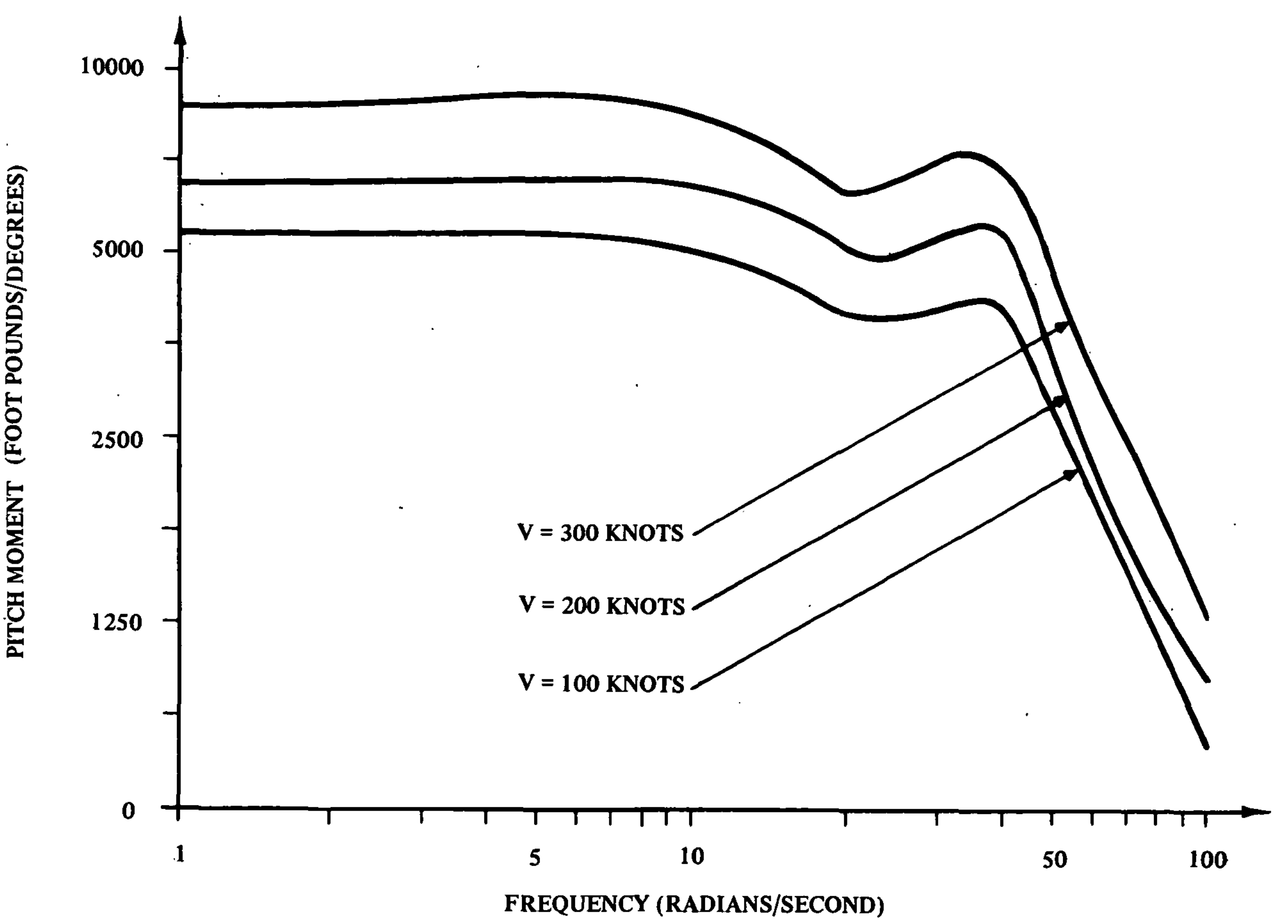


FIGURE 81 RSRA ROTOR PITCH MOMENT FREQUENCY RESPONSE

TABLE XIV
ROTOR RESPONSE TRANSFER FUNCTIONS

OUTPUT/INPUT (UNITS)		AIRSPEED	TRANSFER FUNCTION
Pitch Moment / Pitch Cyclic	$\left(\frac{\text{Ft Lb}}{\text{Deg}}\right)$	100 Knots	$\frac{108.8 \times 10^6}{(S^2 + 46S + 1444)(S + 12)(S + 50)}$
		200 Knots	$\frac{165.6 \times 10^6}{(S^2 + 46S + 1444)(S + 12)(S + 50)}$
		300 Knots	$\frac{293.9 \times 10^6}{(S^2 + 46S + 1444)(S + 12)(S + 50)}$
Roll Moment / Roll Cyclic	$\left(\frac{\text{Ft Lb}}{\text{Deg}}\right)$	100 Knots	$\frac{120.8 \times 10^6}{(S^2 + 46S + 1444)(S + 16)(S + 50)}$
		200 Knots	$\frac{143.6 \times 10^6}{(S^2 + 46S + 1444)(S + 16)(S + 50)}$
		300 Knots	$\frac{247.0 \times 10^6}{(S^2 + 46S + 1444)(S + 16)(S + 50)}$
Thrust / Collective	$\left(\frac{\text{Lb}}{\text{Deg}}\right)$	100 Knots	$\frac{3718(S + 7)(S^2 + .5S + 625)}{(S + 12)(S^2 + 24S + 400)}$
		200 Knots	$\frac{7119.9\ (S + 7)\ (S^2 + .5S + 625)}{(S + 12)(S^2 + 24S + 400)}$
		300 Knots	$\frac{13226.0\ (S + 7)(S^2 + .5S + 625)}{(S + 12)\ (S^2 + 24S + 400)}$

ROTOR FEEDBACK ANALYSIS

Root locus analyses of several feedback systems were performed and the error plus integral of error system shown in Figures 82 and 83 was selected. This system provides low error and acceptable frequency response characteristics. Due to the similarity of the transfer functions at all airspeeds, the ratio of the gains on error and integral of error (equivalent zero location) did not vary with airspeed. The increased sensitivity of the rotor at higher airspeeds did require that the feedback gains be reduced to keep the system gain constant.

The rotor control system contains many components, each of which has its own dynamic characteristics. In order to determine the available stability margin of the feedback control systems, Nyquist plots were made and the system gain and phase margins determined. Figures 82 and 83 also show the Nyquist plots for the pitch and collective controls. The roll control is similar to the pitch control and is not shown separately. Sufficient stability margin is available with this system to allow reasonable confidence in the practicality of the concept. Careful choice and design of the system components will keep the stability margin from being used up and maintain a high level of performance.

This feedback control system was programmed on the hybrid computer along with the S-67 rotor. Sensor and actuator dynamics and nonlinearities were simulated. Digital computer data rate was also simulated to determine its effect on the rotor control system performance. The feedback gains were optimized on the simulation to achieve maximum performance. The criteria used for optimization were minimum system response time and minimum number of overshoots. It was discovered that slightly higher gains could be tolerated on the simulation than indicated by the root locus plots but otherwise good correlation was obtained. Transient responses to pitch moment command inputs are shown in Figure 84 and show generally good performance. Less than satisfactory response was found at 300 knots. This is attributed to the onset of stall and the decreased angle of attack stability at that speed. The addition of delta three coupling (pitch-flap) to the rotor as a stabilizing factor was briefly evaluated. The rotor become more stable at high speeds with increased delta three but a thorough evaluation was not conducted at this time.

An evaluation of the effects of the dynamic characteristics of the system components revealed the latitude available in the choice and design of these components. Parameters varied during this study were: sensor lag, servos hysteresis, computer program cycle time, actuator lag, and actuator hysteresis. Hysteresis values of reasonable magnitude (1-2% of total control travel) had little or no effect on system performance. Similarly, actuator lags of usual proportion had no effect. The major system degradations came from sensor lag and computer cycle time. Figure 85 shows the effects of these parameters on rotor stability. As was expected, high computer speeds do little to improve performance as do very low sensor lag times. A reasonable limit on these variables is a computer program cycle rate of greater than 5 per rotor revolution and a sensor lag time constant less than .15 seconds.

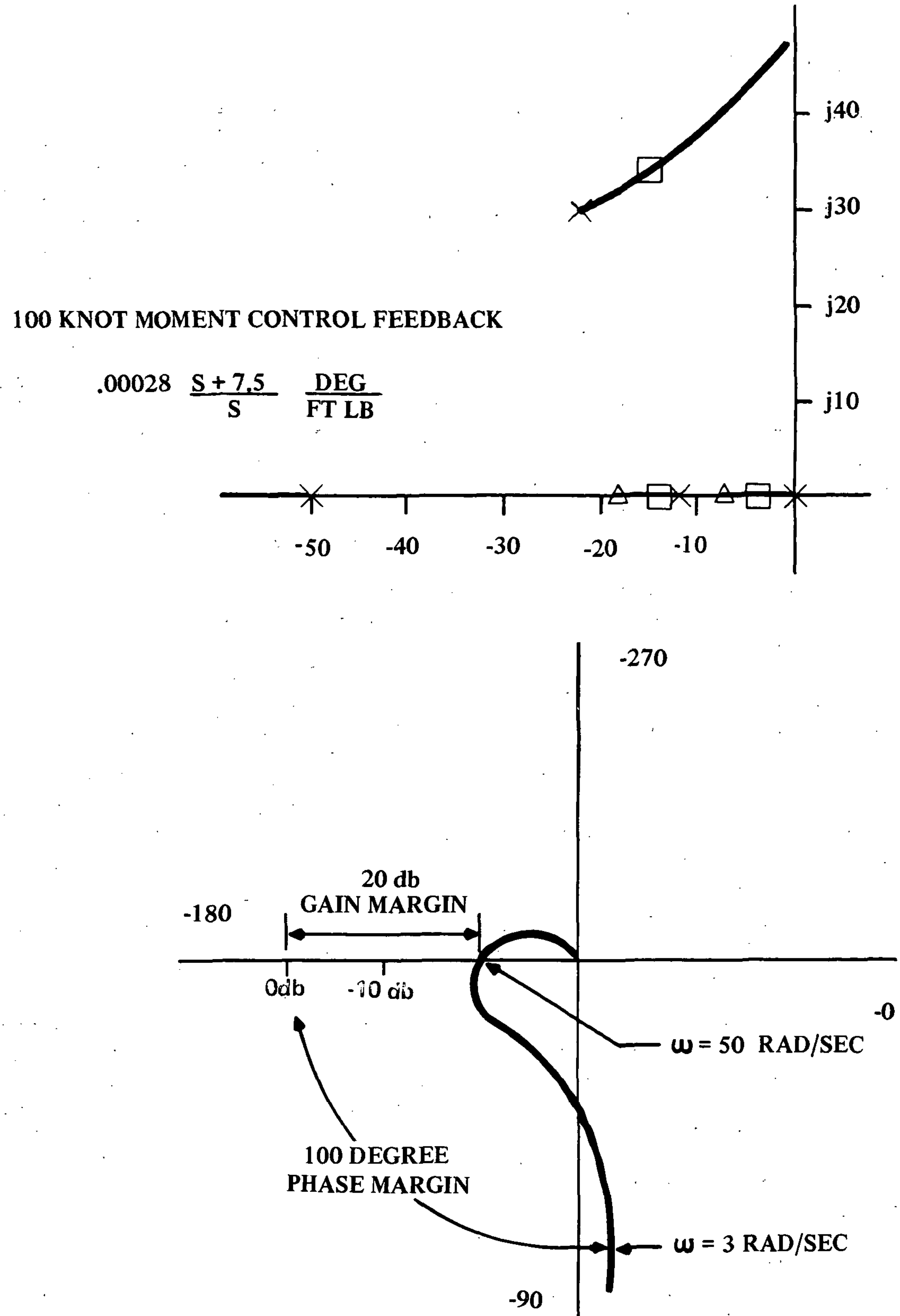


FIGURE 82 PITCH MOMENT FEEDBACK SYSTEM

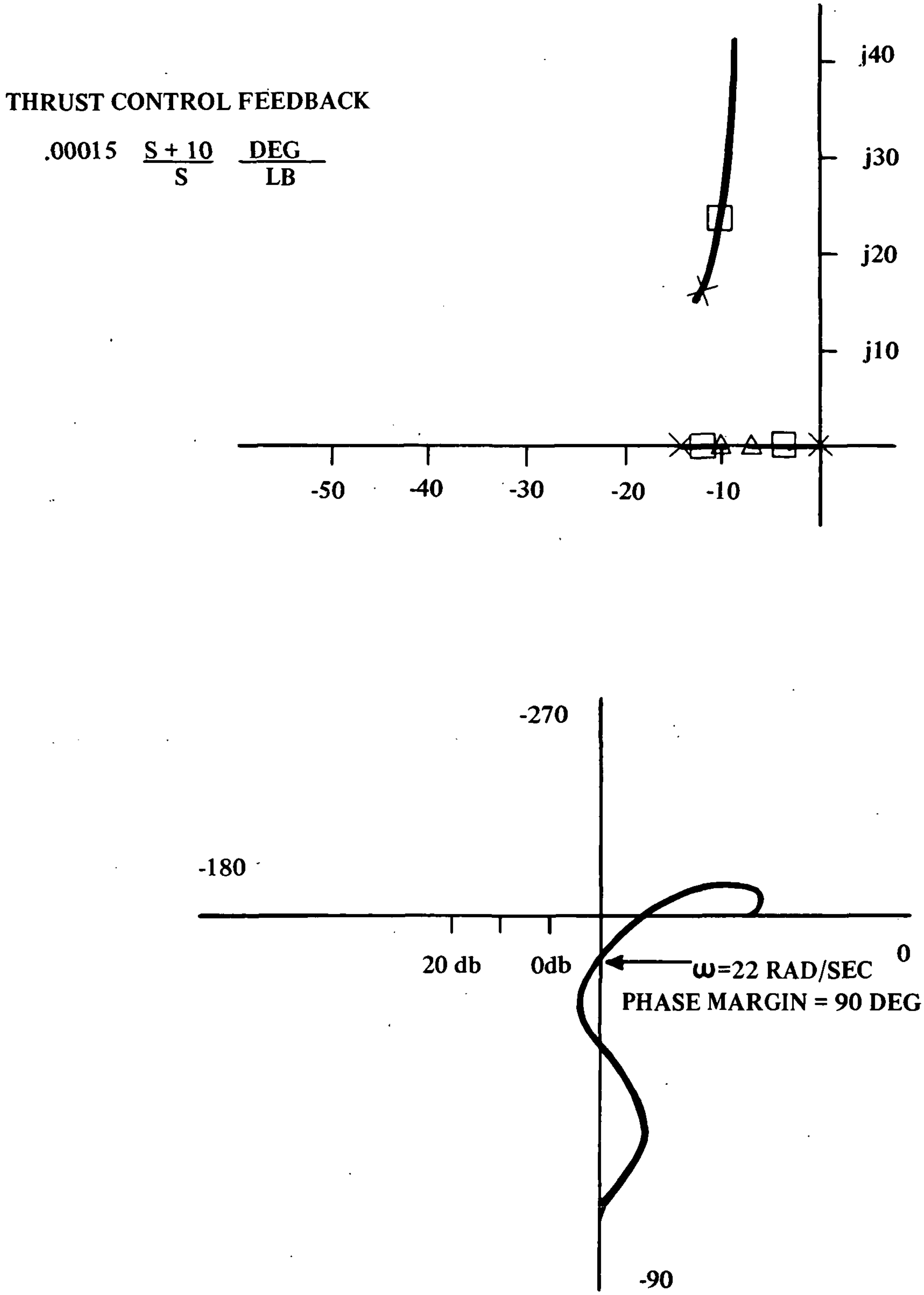


FIGURE 83 ROTOR THRUST FEEDBACK SYSTEM

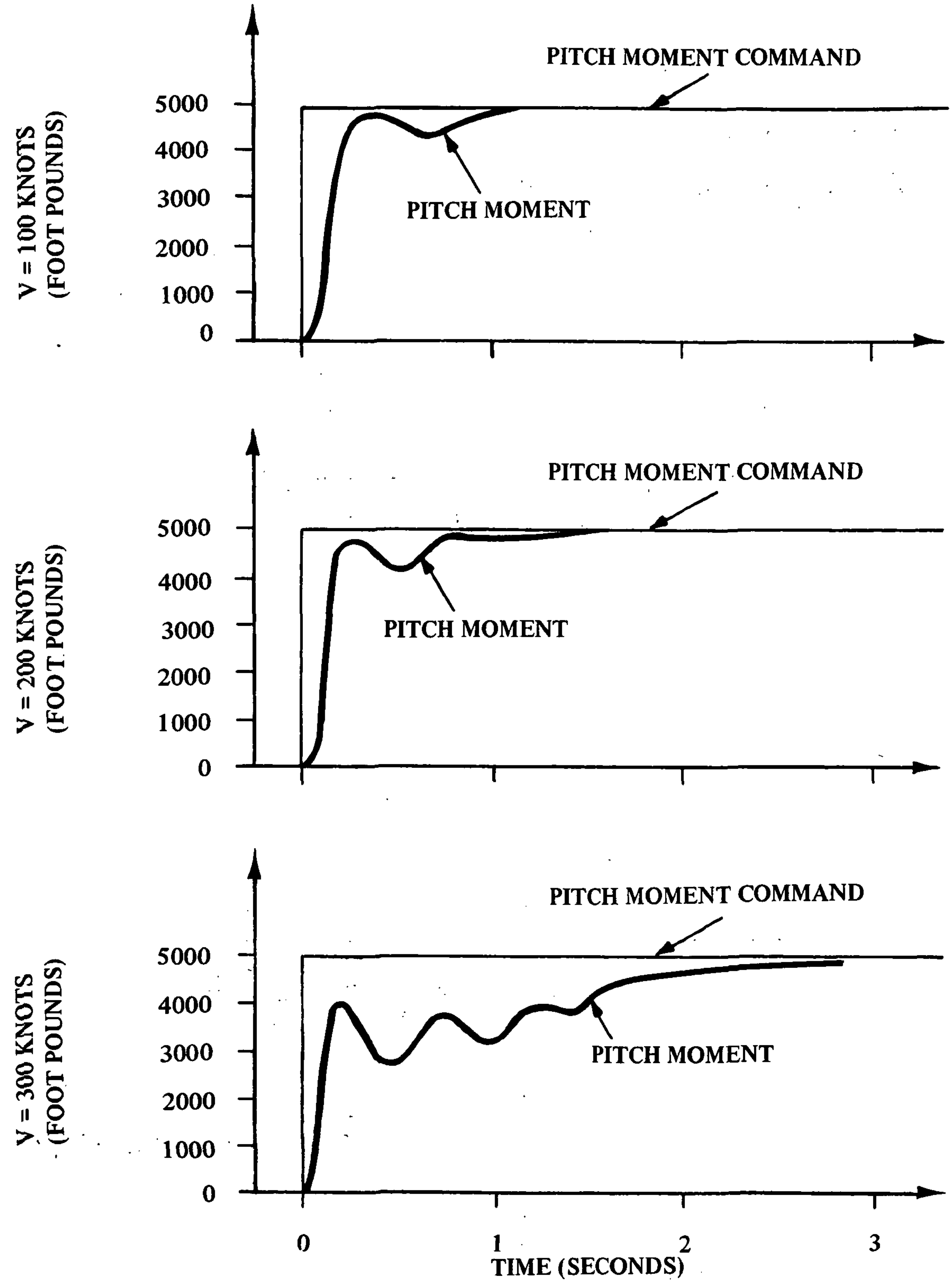


FIGURE 84 ROTOR RESPONSE TO PITCH MOMENT COMMAND

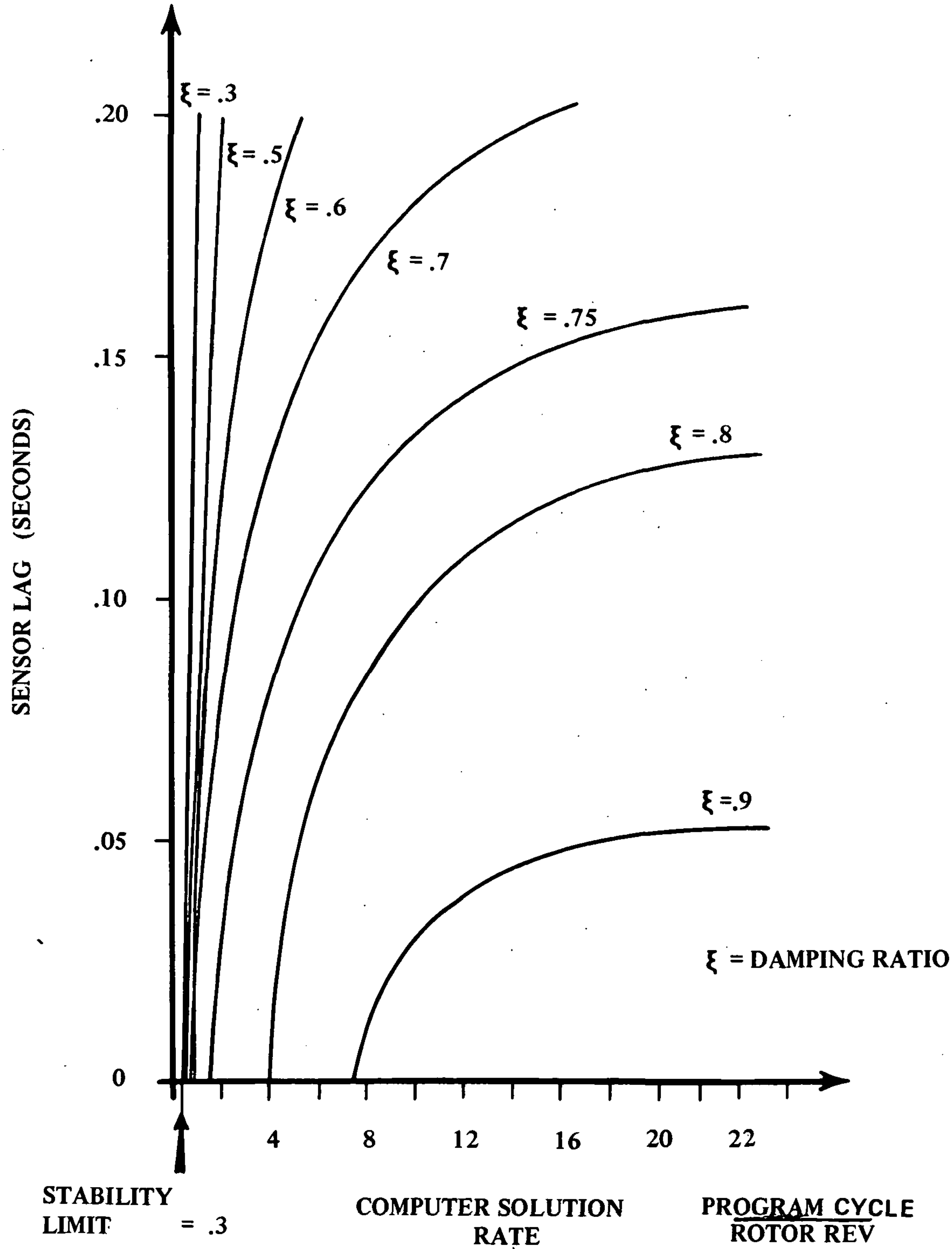


FIGURE 85 SENSOR LAG AND COMPUTER RATE EFFECTS ON SYSTEM STABILITY

The RSRA Aircraft has been designed to allow a wide range of aircraft capability to test and simulate various aircraft configurations and rotor systems. The wing and drag devices were sized without regard to the particular limits of the S-67 rotor system on the basic aircraft. The capability presented here is that of the aircraft as a rotor test vehicle and particular rotor limits are not imposed.

Range of Disc Loadings

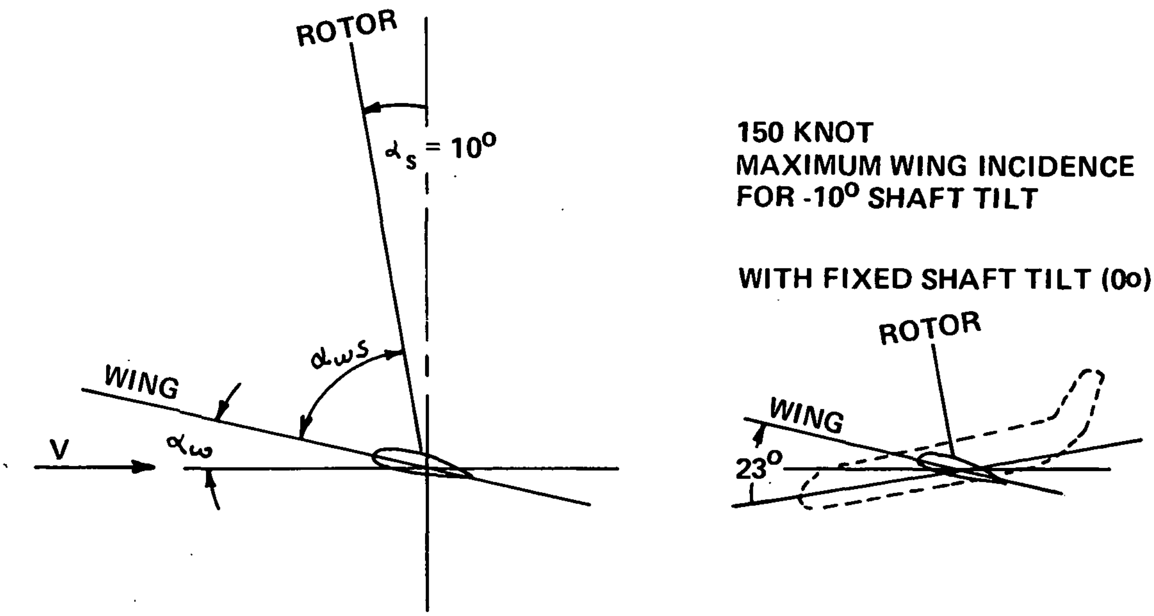
The range of disc loadings that can be tested by various rotor sizes between 100 and 200 knots is from 5.5 to 20 psf as discussed in the section on Range of Rotor Sizes. While these ranges can be achieved, the high disc loading rotors would not be capable of meeting the RSRA hovering missions with the current shaft power engines.

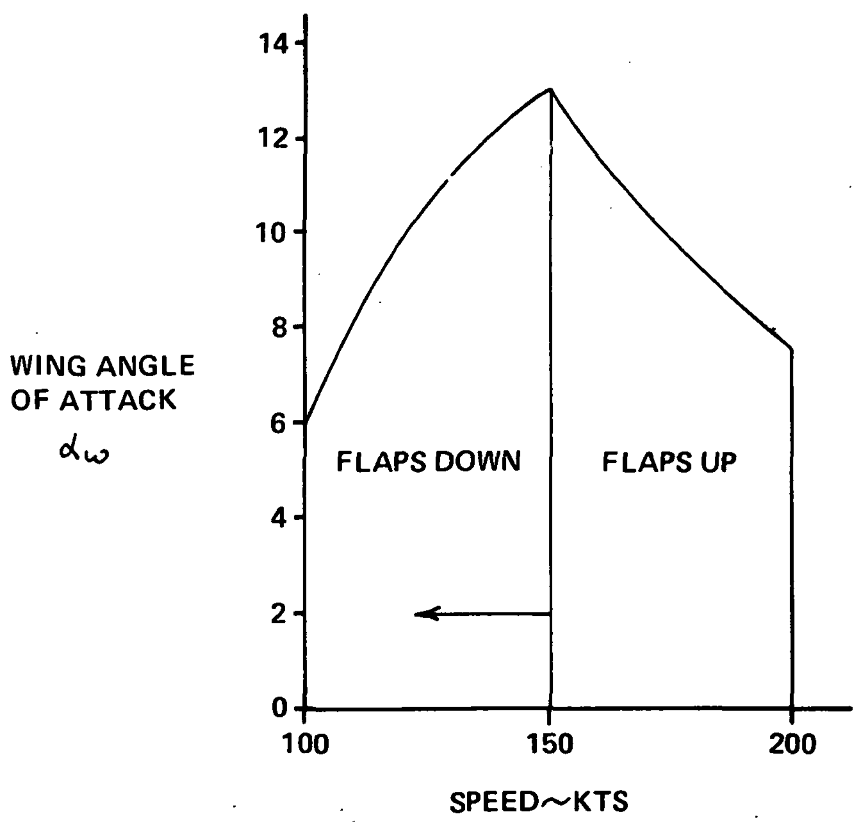
Shaft Angle Simulation, Wing Angle Requirements

In order to provide lightly loaded rotor performance and full gross weight autorotation at various shaft angles, a study was undertaken to establish the wing incidence requirements to allow for these conditions. At zero rotor lift, the wing unloads the rotor completely without flaps down to 150 knots and with flaps below 150 knots. For this study, it was assumed that the flaps were capable of unloading the rotor completely to 100 knots, the NASA/Army design goal. A capability to test rotors at a -10° shaft angle was assumed to be the limit of any desirable rotor test.

Figure 86 shows the rotor and wing angles with respect to the free stream velocity at the 150 knot flaps up point with the desired -10° shaft tilt. The angle between the shaft and the wing chordline is defined as α_{ws}. The graph at the bottom shows the wing angle of attack required to fully unload the rotor from 100 to 200 knots. The figure shows that the maximum wing angle occurs at the 150 knot point. At this point with the wing angle of attack at 13°, it can be seen that α_{ws} will be a minimum to achieve a shaft angle of -10° across the speed range from 100 to 200 knots. This minimum included angle is 67 degrees. The maximum wing-fuselage angle is 23 degrees, the ten degree nose down attitude of the fuselage plus the wing incidence angle with respect to free stream air.

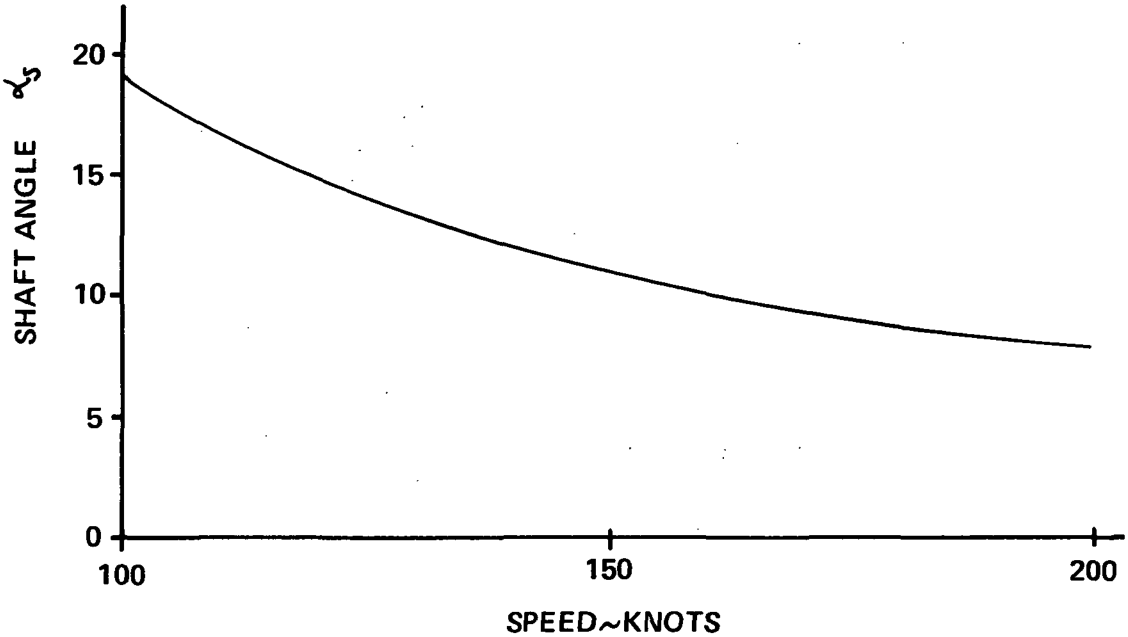
Negative wing angle requirements for the RSRA are defined by the desired autorotative capability at full gross weight. In autorotation, the rotor thrust is obtained by positive rotor angle of attack with near-zero collective. The rotor angle of attack must increase to maintain thrust as speed drops. For autorotation from 100 to 200 knots, the rotor shaft angles were calculated by a linearized analysis and are shown in Figure 87 . The highest rotor shaft angle required is 19° at the 100 knot condition.

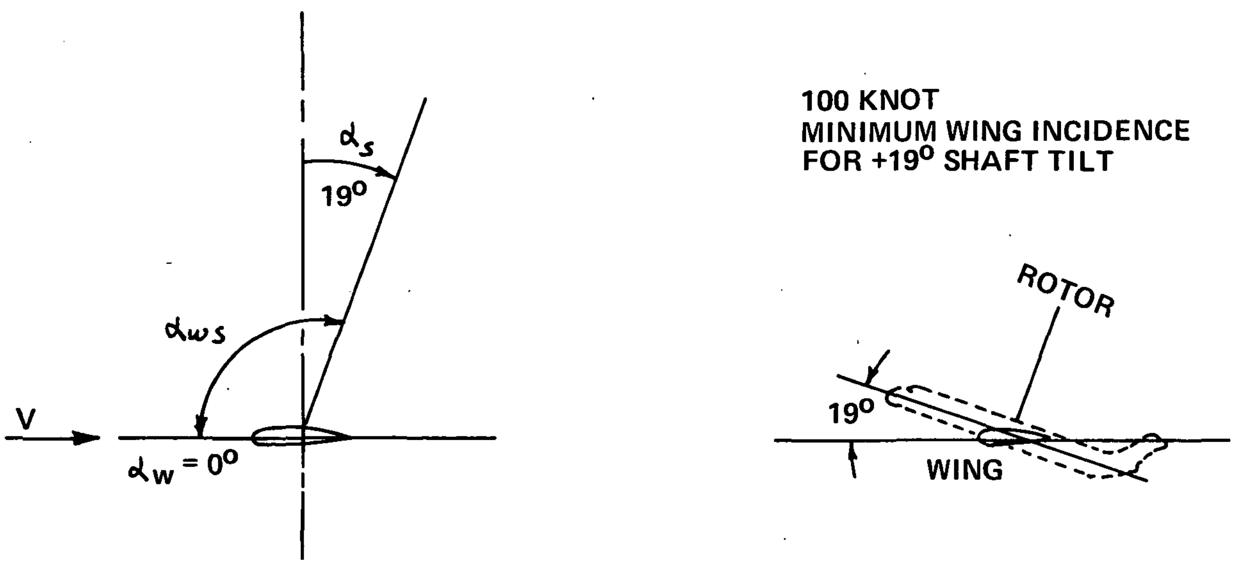




MAXIMUM WING INCIDENCE REQUIREMENTS

FIGURE 83





AUTOROTATION WING TILT REQUIREMENTS

FIGURE 87

In order to autorotate with the rotor carrying full gross weight, the wing must be unloaded. This is accomplished by placing the wing at zero angle of attack. The sketch at the bottom of Figure 87 shows the rotor/wing angles with respect to the stream velocities, at the 100 knot condition. α_{ws} is again defined as the angle between the rotor shaft and wing chord line. From the two figures, it can be seen that the maximum value of α_{ws} is at the 100 knot condition. With the rotor incidence at zero degrees, α_{ws} is 109^{o} and the wing/fuselage angle is -19^{o}.

Wing/Rotor Interference

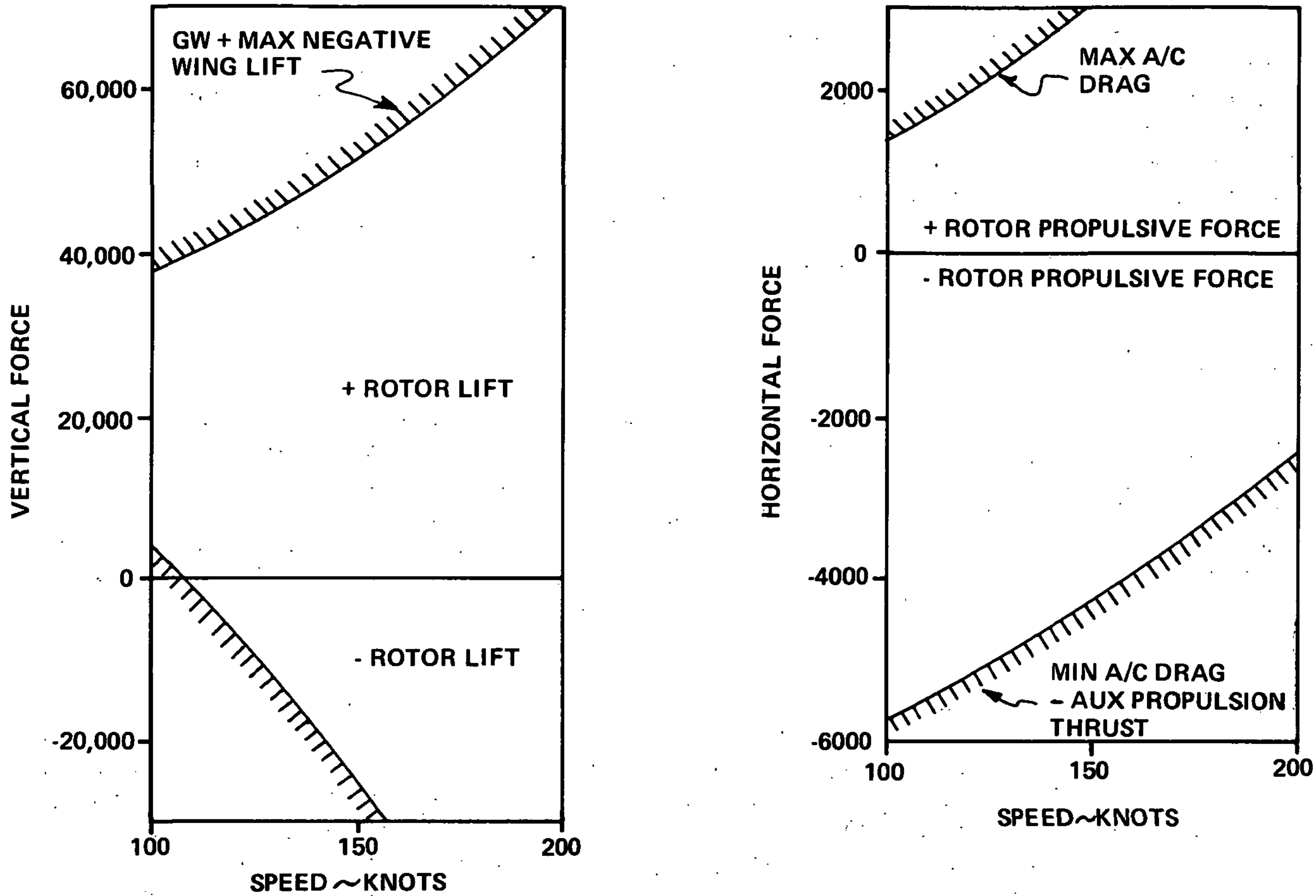
With variable wing incidence, most of the rotor influence on the wing can be compensated for by varying the wing incidence. The interferences of the wing on the rotor have been found to be significant only for highly unloaded rotors. Although interesting for a test aircraft, this condition is not normally a part of helicopter operating spectrums and is therefore relatively unimportant. Interference effects are discussed in more detail in Appendix C of this report.

Performance Mapping

The aircraft's large wing has been designed to fulfill the requirement to support the full gross weight of the aircraft at 150 knots, sea level, standard conditions, in a clean, unflapped configuration. With this amount of wing available plus the inclusion of wing high lift devices, the capability to provide a wide range of rotor loading conditions is available. Figure 88 shows the capability of the aircraft to react rotor forces and/or unload the rotor at a given cruise speed. With the negative wing angles required by autorotation, the wing has the capability to produce sufficient negative lift to load up conventional rotors to their upper stall limits.

The drag device and the auxiliary propulsion provide a wide range of rotor propulsive forces which can be reacted by the aircraft at any given speed. Figure 88 also shows the capability range provided the aircraft with the drag brakes 100 percent deflected and with the minimum aircraft drag and the auxiliary thrust on.

This analysis illustrates how the RSRA airframe can generate a complete range of reactive forces for testing rotors over a large operating spectrum. Rotors can be tested from close to zero rotor lift to their stall limit, and from maximum propulsive force to full autorotation.



AIRCRAFT PERFORMANCE -- HELICOPTER SIMULATION
CAPABILITY OF THE AIRCRAFT TO REACT ROTOR FORCES

FIGURE 88

CONTROL DURING SIMULATION

The RSRA can be controlled in such a manner as to simulate the fuselage characteristics of most helicopters. The control of the drag brakes, engine thrust, flaps, rudder, ailerons and stabilator allows full control of 5 of the possible 6 degrees of freedom. This allows the flexibility to vary the fuselage aerodynamic characteristics through the computer by feeding back the proper signal to the proper control. For example, a change in pitch damping from the RSRA value may be simulated by feeding pitch rate to the stabilator.

The digital computer will be responsible for control of the RSRA during the helicopter simulation experiments. The pilot, through either his control inputs or through pre-programmed computations, will conduct the experiments. The co-pilot will monitor the aircraft responses. Any copilot input which overrides a computer input will cause an interrupt, terminate the simulation run, and revert the RSRA to normal control.

(1) PREPROGRAMMED TESTING

A considerable savings in flight time may be realized by allowing the computer to command the aircraft and rotor parameters during the data runs. The computer will select and set the many parameters much more rapidly than can a pilot. Further, the accuracy of the data should be more consistent between flights. To accomplish the testing, the pilot will bring the aircraft to the desired starting airspeed and altitude. He will then set the drag brake and engine thrust at the proper predetermined setting. He will set the flaps at half travel and adjust the wing incidence to bring the rotor thrust near the desired level. The half flap setting will allow the computer to command decreased lift as well as increased lift. Starting the computer program at this point will cause the feedback loops to close and bring the fuselage and rotor parameters to the desired trim values.

The RSRA hybrid computer simulation was used to evaluate the preprogrammed testing capability of the RSRA. Figures 89 and 90 show the time responses of commanded changes in rotor thrust and pitch moment respectively. The fuselage parameters and remaining rotor parameters are commanded to remain constant. The control response is, in all cases, smooth and accurate with minimum response of the variables commanded to remain constant.

Performing preprogrammed maneuvers is also possible and this capability allows the maneuvers to be repeated accurately and consistently. Figure 91 shows the simulation of a preprogrammed symmetrical pull up from level flight. The rotor forces and moments are commanded to remain constant during the maneuver. For these tests the computer is programmed as a high gain autopilot. Changes in flight path parameters can be commanded in sequence and at defined rates. The RSRA will follow the commanded changes and perform the maneuver. The pull up maneuver of Figure 91 was performed by engaging the system and allowing the parameters to trim, then commanding a pitch attitude change.

(2) MODEL-FOLLOWING

The model following concept has been proven and well documented in recent years. Several programs such as the Air Force - Cornell Total In-Flight Simulator have been untertaken in this area and the technology is fairly well developed. During flight in this mode, the pilot's electrical inputs are sent to the computer where they are shaped and sent to the model. The model outputs are compared to the aircraft outputs and the error is shaped and sent to the appropriate controls. A specific model-following scheme was not studied but the capability of an on board digital computer complete with the control capability of the RSRA should provide adequate model-following performance using one of the already developed model-following techniques.

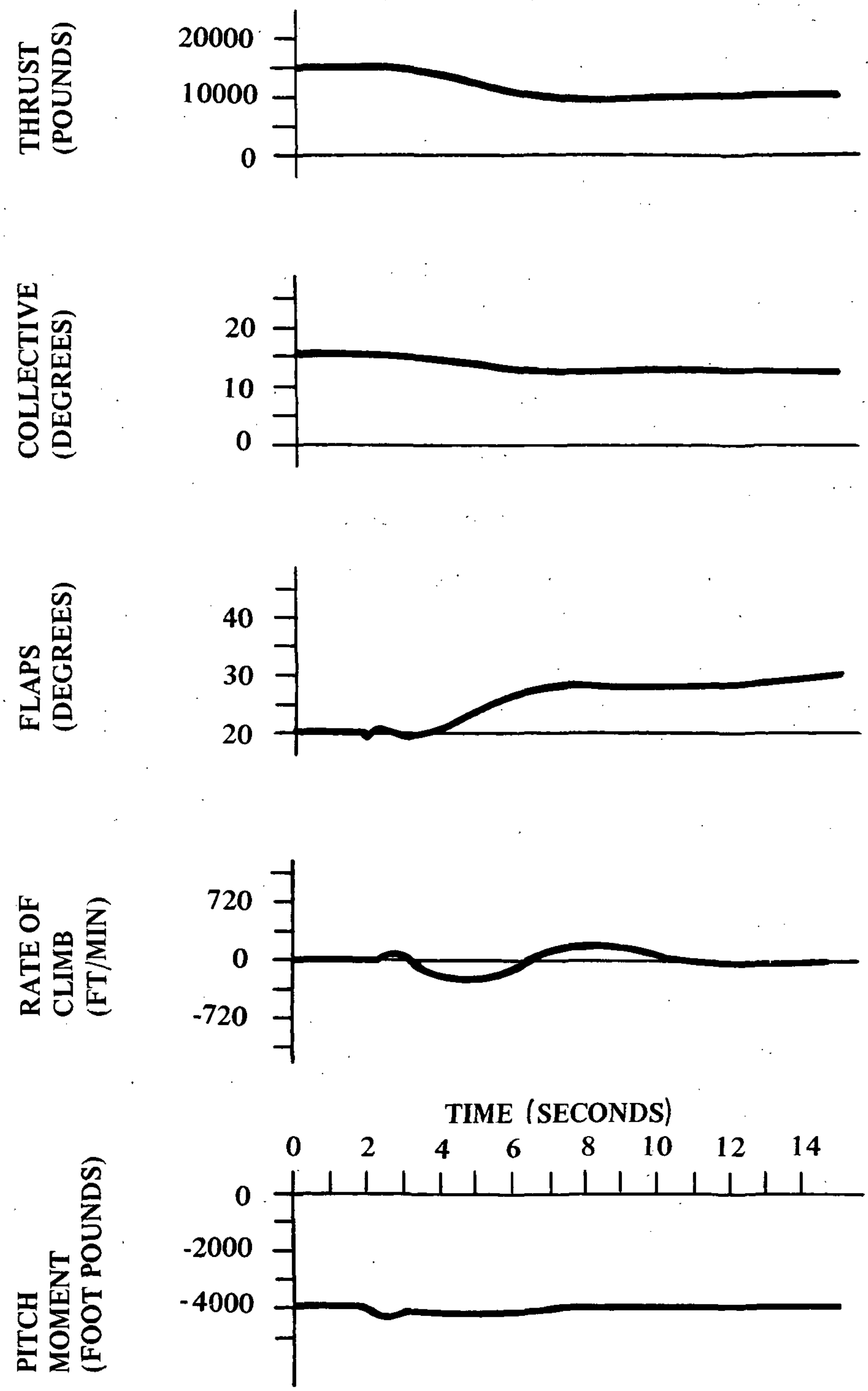


FIGURE 89 RSRA RESPONSE TO COMMANDED CHANGE IN ROTOR THRUST

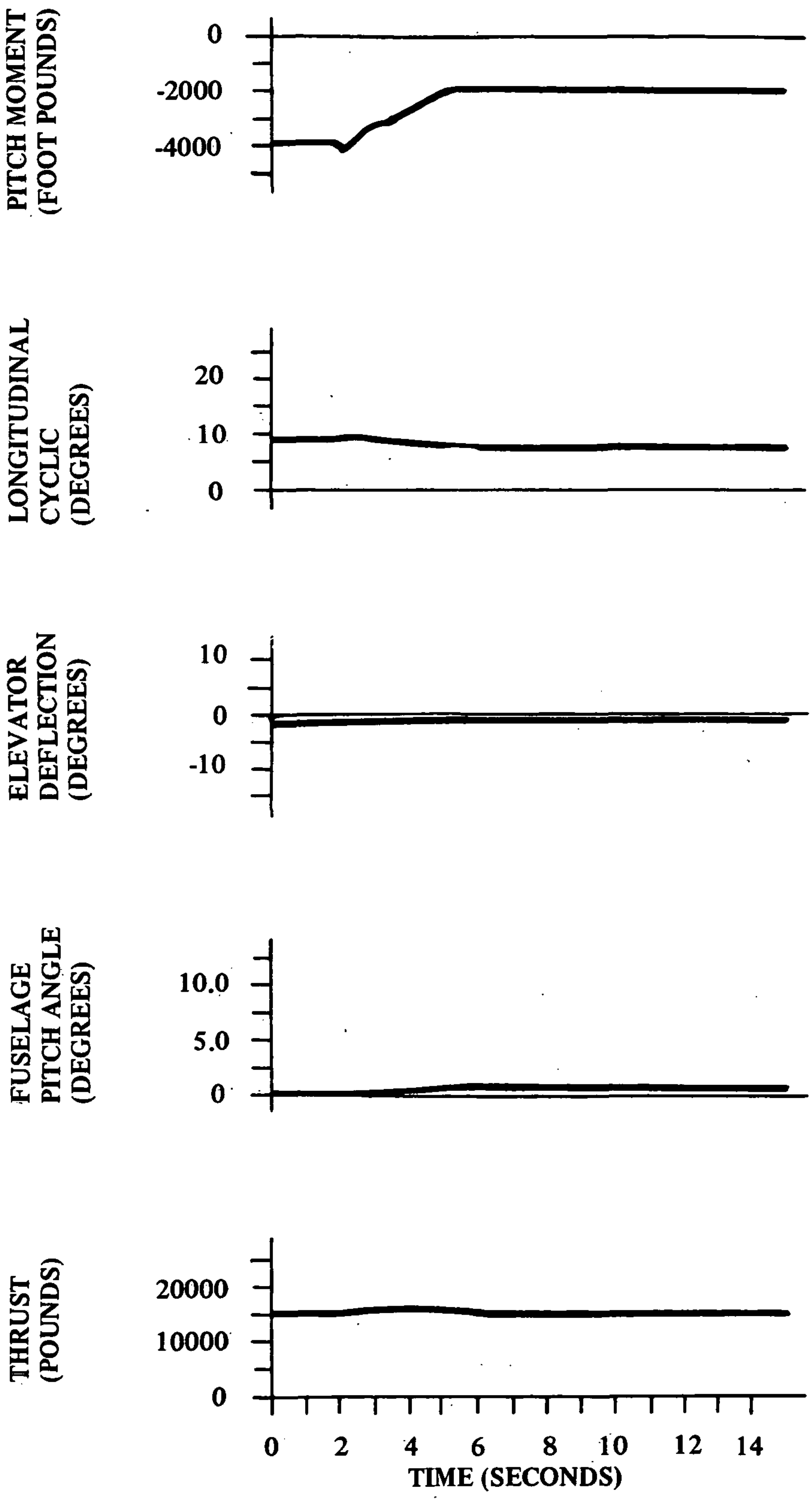


FIGURE 90 RSRA RESPONSE TO COMMANDED CHANGE IN PITCH MOMENTS

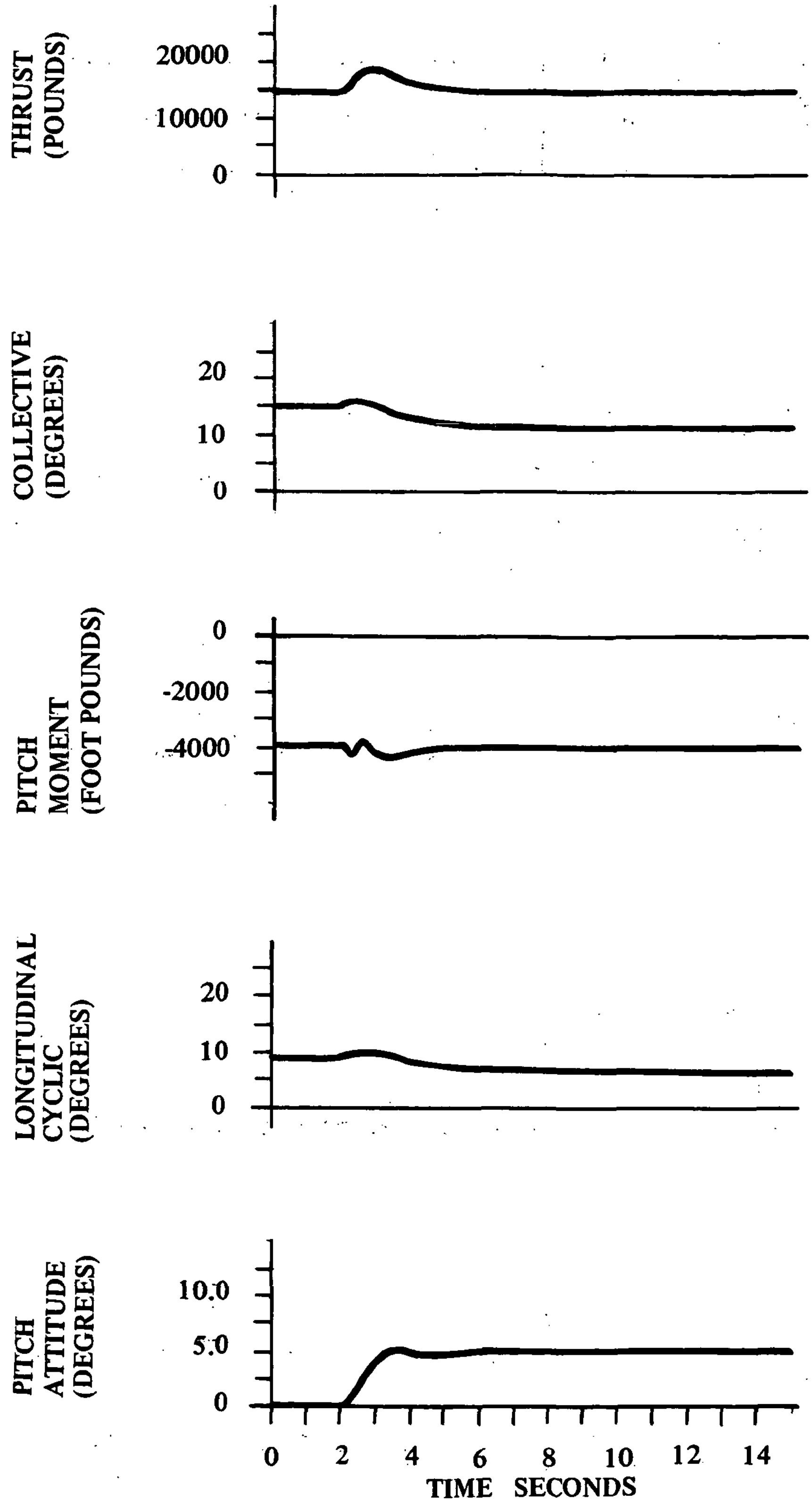


FIGURE 91 RSRA PULLUP MANEUVER

PROGRAM RISK ASSESSMENT

A primary goal during the predesign study was to assess the RSRA program risk. This is important because the RSRA does include various new and unique features not normally included in conventional rotary wing aircraft. To do this, the unusual features of the final RSRA configuration were identified and a risk assessment performed on each one. Five specific areas were investigated:

- The load cell mounting of the main gearbox
- Rotor/Airframe Dynamic compatibility when many different rotors are installed on one airframe
- The basic flight control system
- The need for, and feasibility of, a rotor feedback control system
- The crew escape system

Although all of these areas involve development, it is concluded that none has sufficiently high risk to warrant postponement of the RSRA aircraft development program. Based upon past experience and present knowledge, each can be incorporated into the RSRA without unduly adding to the program risk.

Load Cell Mounting of the Main Gearbox

The rotor load cell force measuring system uses four vertical and three horizontal load cells to measure all rotor forces and moments. These cover all load paths between the gearbox and the airframe, so that all loads can be measured. The load cells are mounted through spherical bearings so that only axial loads will be transferred through each load cell. The units selected are commercially available, and are designed to operate in fatigue applications. They can withstand more than 10^8 fully reversed cycles without failure.

Although RSRA will probably be the first aircraft to mount the complete rotor transmission system on load cells, this is not because of any technical problems or risks associated with the concept. RSRA is the first aircraft that is specifically being designed to measure forces and moments in flight to the high accuracy levels that require this type of hardware. Sikorsky has long used similar load cell rotor force and moment measuring systems on ground test stands. The Sikorsky main rotor Whirlstand can test rotors with over 100,000 pounds of lift. It has been in operation for over fifteen years, using load cells to measure rotor thrust, horizontal force, and torque.

Rotor/Airframe Dynamic Compatibility

Six advanced rotor systems were considered as representative of the types of rotors which might be tested on the RSRA. These were in addition to conventional helicopter and compound rotors, and included a six-bladed variable geometry rotor, the variable diameter rotor, rigid coaxial, jet flap, variable twist, and slowed rotors. Considering the large variations possible in blade number, radius, and tip speed, they cover a wide band of principle blade passage frequencies. It would be impossible to design an airframe so that all modes of vibration will never be resonant with all possible vibratory excitation frequencies from these rotors.

It is possible to dynamically tune the airframe to accept three, four, five, and six bladed rotors if their RPM bands are within reasonable limits. This will permit testing of these rotors without the use of either active or passive vibration suppression systems. Other rotors, such as two bladed rotors, slowed rotors which operate over a wide band of rotational speeds, and rotors with unusual RPM operations, will require some type of vibration suppression system.

Because the airframe can be dynamically tuned, it is suggested that the RSRA be developed without an isolation system. If a full universal capability is indeed required, an active vibration system could be developed as a parallel effort. During the predesign study a full active system has been conceptually designed which will accept all types of rotors operating over an extensive RPM range. This system would replace the main gearbox to airframe load cell mounting hardware and can be used to also serve as a rotor force and moment measuring system.

Airframe Dynamic Tuning - Consider the five-bladed compound rotor operating at forward speeds up to 300 knots, the six-bladed variable geometry rotor, and the four-bladed variable diameter rotor. These can be accommodated on the RSRA without the need for vibration suppression devices through structural tuning of the airframe. This will also permit the testing of other three, four, five, or six bladed rotors operating at similar rotational speeds.

Two tuned airframe configurations are recommended. The first will accommodate the compound and variable diameter rotors. The second will accommodate the variable geometry rotor. To tune the airframe, the transmission pitch, transmission roll, second lateral, and second vertical bending modes must be controlled. Experience indicates that these modes are uncoupled, and their locations are controlled by the stiffness of different portions of the airframe. The transmission pitch mode is controlled by the stiffness of the top of transmission support frames. The transmission roll mode is controlled by the stiffness of the sides of these frames. The second lateral and second vertical bending modes are controlled by the lateral and vertical bending stiffness of the aft fuselage and tail cone, respectively. The basic vehicle will be designed to locate the modes at the lower of the two required positions. The frequencies of these modes can then be increased as required through the addition of material.

The feasibility of shifting the location of fuselage modes has been demonstrated during full-scale ground tests at Sikorsky Aircraft. In addition, three-, five-, and six-bladed rotors have been successfully flight tested on a single aircraft, the S-61F (NH-3A) high speed research aircraft.

Active Transmission Isolation - Airframe tuning provides the capability of testing the compound rotor, variable diameter rotor, variable geometry rotor, and any other rotor system whose primary excitation frequencies fall within the bands produced by these rotors. Tuning for other rotor systems whose frequencies fall beyond these bands, particularly slowed rotors, can be expedited through use of a variable tuning device.

Active transmission isolation can provide full wide band tuning for RSRA. Static and transient displacements are actively controlled. Spring rates can be made as low as required to provide wide band isolation. The proposed configuration of the Sikorsky active rotor balance vibration suppression system uses seven self-contained hydropneumatic actuators (isolators) to decouple the pitch and roll modes and thus provide independent focusing of the isolation system. Circular tracks are provided so that focusing can be varied easily. This variation can be accomplished independently in pitch and roll.

Analyses have substantiated the ability of this system to isolate the airframe from all rotor forces and moments simultaneously while providing accurate measurement of principal rotor forces.

Basic technology required for development of the active rotor balance/vibration suppression system has been demonstrated. Full-scale laboratory experiment of the hardware acting also as a rotor balance is scheduled.

A full-scale active transmission isolation system has been fabricated and ground tested under contract to the U.S. Army. It has the ability to provide an overall 70% reduction in vibration to vertical and inplane forces at the particular blade passage frequency of interest. Of greatest significance to RSRA is the wide band characteristic achievable with this system.

The feasibility of using active isolator units as load sensing devices was also demonstrated successfully during a recent NASA-supported effort. The accuracy of measuring steady loads was found to be within a band of ±1 percent of applied load about a linear bias that can be removed through calibration.

Calibration of the total system as a rotor balance will take place later this year under a NASA/Army-supported program that is currently under way. The system, which contains three active hydropneumatic units, will be instrumented and installed on a CH-53A aircraft. Instrumentation will include hydraulic pressure transducers, inplane drag strut load cells, and transmission-mounted accelerometers required for transient and vibratory measurement.

Flight Control Systems

A combination electrical and mechanical flight control system has been selected for RSRA to provide testing versatility with low cost and risk. The test pilot's controls are electrical and are separated from the safety pilot's mechanical controls. With this system the test pilot's controls can be shaped and varied as required, yet the safety pilot has a completely mechanical system which can override pilot's system. The safety pilot is therefore responsible for monitoring aircraft status and returning the aircraft to normal flight from any test condition.

This type of control system for RSRA is a relatively low risk approach that has counterparts in several current experimental and production aircraft. The major programs that use an electrical/mechanical system are the Cornell TIFS, CH-54B, and NASA V107.

The two types of electrical control systems currently employed are the mechanical following system and the mechanical reversion system. The primary advantage of the mechanical following system, such as that in the RSRA, is its set of mechanical controls, which are linked to the control surface at all times. The U.S. Army/Sikorsky CH-54 has an electric stick at the rear-facing seat configured in much the same fashion.

Many of the control system components of the RSRA already exist on other Sikorsky helicopters and are adapted to the RSRA. The development risk for the remaining components is low due to the conventional design. The primary servos, auxilliary servo, and mixing unit of the RSRA are the Sikorsky S-61/S-67 components. They are installed in the Navy SH-3, the Air Force CH-3, the commercial S-61L and N, and the S-67 Blackhawk. No additional development is required to adapt these components to the RSRA. The mixing unit will have to be modified to allow control of rotors with different control phasing, but the technology exists and is well developed.

RSRA uses control integration units of new design. Success in building similar units for the XC-142, F-111 and other variable geometry aircraft makes the development of these units a low technical risk.

The control surface actuators are similar in concept to the actuators currently in use on many aircraft of variable types. They consist of a high speed, limited authority series servo and two low speed, full authority, series servos. The high speed servo is similar to the SAS input of the auxiliary servo and the low speed servo is similar to the trim actuator of the H-53 and H-54B. The development of these actuators is therefore low risk.

A prototype FAS (Force Augmentation System) system has been developed and flown on the CH-53 and S-67 Blackhawk. A production version will be developed for the RSRA. This development will involve only a repackaging effort and will be a low risk effort.

Rotor Feedback Control System

Rotor feedback control is not required for the RSRA to perform its basic research function. It would, however, add to the testing capability of the aircraft because it could provide:

- An automatic trim feature to expedite testing at a predetermined loading condition, independent of the normal trim characteristics of the test aircraft.

- The ability to dynamically simulate the fuselage/wing characteristics associated with the rotor design being tested.

- A flexible system that can provide any of the test rotors with the ability to test various control and feedback schemes. Some of these that might be desired for the next generation high speed helicopter include gust alleviation, adaptive control configurations, and modal suppression.

State feedback to accomplish model-following is not new in terms of helicopter technology. Several helicopters are currently being used as variable stability systems. Rotor feedback has not been done as extensively, and Sikorsky currently has a NASA/Army contract to begin to investigate this area.

The objective of this separate rotor/vehicle state feedback contract is to:

(1) Establish, using a CH-53, the feasible bandwidth of rotor state control by means of high gain feedback of several possible rotor and vehicle state variables.

(2) Quantify on the CH-53 the gust suppression capabilities of various possible rotor and vehicle state feedback loops.

The first portion of the study will be analytic, with the computer techniques and the system stability being investigated using both linear and nonlinear dynamic programs. The second part of the study involves flight testing. Feedback will be introduced into the helicopter control system through the existing limited authority AFCS. An airborne computer will be used to condition and shape the feedback information before it is routed to the AFCS servos.

The question of concern in regard to rotor feedback is one of degree of tracking accuracy required rather than one of whether or not the job can be accomplished. This test program will help to answer these questions.

The RSRA will algebraically resolve rotor load cell data into the six rotor forces and moments. These force and moment signals can be fed back into the control system to provide automatic control inputs to trim the rotor or to

actuate the vehicle fixed wing controls.

The predesign study has shown that good response characteristics can be obtained with the proposed aircraft and control system. Analytic studies performed on a PDP-10 hybrid computer with the full RSRA aircraft characteristics have shown that tight response can be obtained for a standard articulated rotor. The hybrid analysis has a rotor blade element solution and uses fuselage aerodynamic data extrapolated from previous compound helicopter wind tunnel data. The full control shaping and feedback network has been programmed on the hybrid in order to fully assess the stability situation. The tight response available from the system can be achieved with practical gains and will operate within nominal authority limits.

To summarize, rotor feedback control is not required for the RSRA but it would add to its testing versatility. With the preliminary results from the predesign study, and after the separately contracted flight tests of a similar system are completed, it should be possible to include a feedback capability in the RSRA without undue risk.

Crew Escape System

The RSRA uses a crew escape system which combines rotor blade severance, canopy separation, and upward crew extraction. It is not considered to be a high risk area since it is the combination of well proven fixed wing type escape systems with the Sikorsky demonstrated rotor blade severance system.

Since early in 1970, Sikorsky has been actively involved in a program to develop a reliable aircrew escape system for helicopters. The upward extraction was desired to provide operation near the ground and to avoid the design complexity of a downward escape system. Upward escape requires removal of the main rotor blades prior to crew extraction, and this can be accomplished by using linear shaped charges around the blade spar. However, two major factors constrained further development of such a system: how to reliably propagate the initiating signal onto the rotating rotor, and how to protect any adjacent aircraft from randomly scattered rotor blades.

The solution to these difficulties has been achieved through the highly successful demonstration of main rotor blade shedding using the sequential main rotor blade severing system. Two demonstrations in December 1971 used a tied-down SH-3 test vehicle. The main rotor blades were sequentially separated in a predetermined direction, three forward and two aft along the longitudinal axis of the vehicle with the main rotor head turning at 203 rpm.

A photo taken from a hand-held camera caught all five blades in flight. The three blade stubs forward hit the road 80 feet ahead of the vehicle within a 5-foot circle. The key to precision of this system is the versatility of the blade sequencing device. It permits any number of blades to be separated singly or in any combination in any direction. When applied to the RSRA vehicle, this will allow the option of simultaneous separation of all blades at once, or sequentially and laterally as proposed for an optional two-stage escape system approach in which the pilot sheds blades only to continue flight as a fixed wing.

The only modification required to adopt this system to any new rotor is the addition of the linear shaped charges clamped around the blade spar, and provisions for the detonating chord.

The blade shedding system is a fully independent system, having no connection with the aircraft's electrical or hydraulic systems. It propagates initiation from the cockpit to the rotor blades through SMDC (Shielded Mild Detonating Chord) with a chemical deflagration rate of approximately 20,000 feet per second. This pyrotechnic system was selected in order to achieve maximum reliability. It is impervious to RF, lightning, and stray voltage. Even gunfire tests with high explosive 20 mm rounds will not cause premature initiation. Deflagration is begun by pilot or copilot activation of percussion primers in the D-rings in the cockpit. Initiation continues through to a sequencing device at the main gearbox, is transferred to the main rotor shaft, and travels out to linear-shaped charges on the rotor blades.

APPENDIX A

Instrumentation Accuracy Study

A procedure was developed to determine the accuracy of the load cell force measurement systems used on the rotor, wing, auxiliary thrust and anti-torque systems. This procedure is shown below for rotor Configuration A. The equations used for the other systems are developed in an analogous manner.

Rotor Load Cell System (Configuration A, Figure 92)

Definitions. - The following definitions are used in the analysis.

H - Long Hub Force (positive aft)
Y - Lat Hub Force (positive starboard)
T - Thrust (positive up)
L - Hub Rolling Moment
M - Hub Pitching Moment
Q - Torque (positive clockwise)

F_{1h} - Horizontal Transducer Force (extreme aft position)
F_{2h} - Horizontal Transducer Force (starboard side)
F_{3h} - Horizontal Transducer Force (port side)
F_{1v} - Vertical Transducer Force (extreme aft position)
F_{2v} - Vertical Transducer Force (starboard side)
F_{3v} - Vertical Transducer Force (port side)

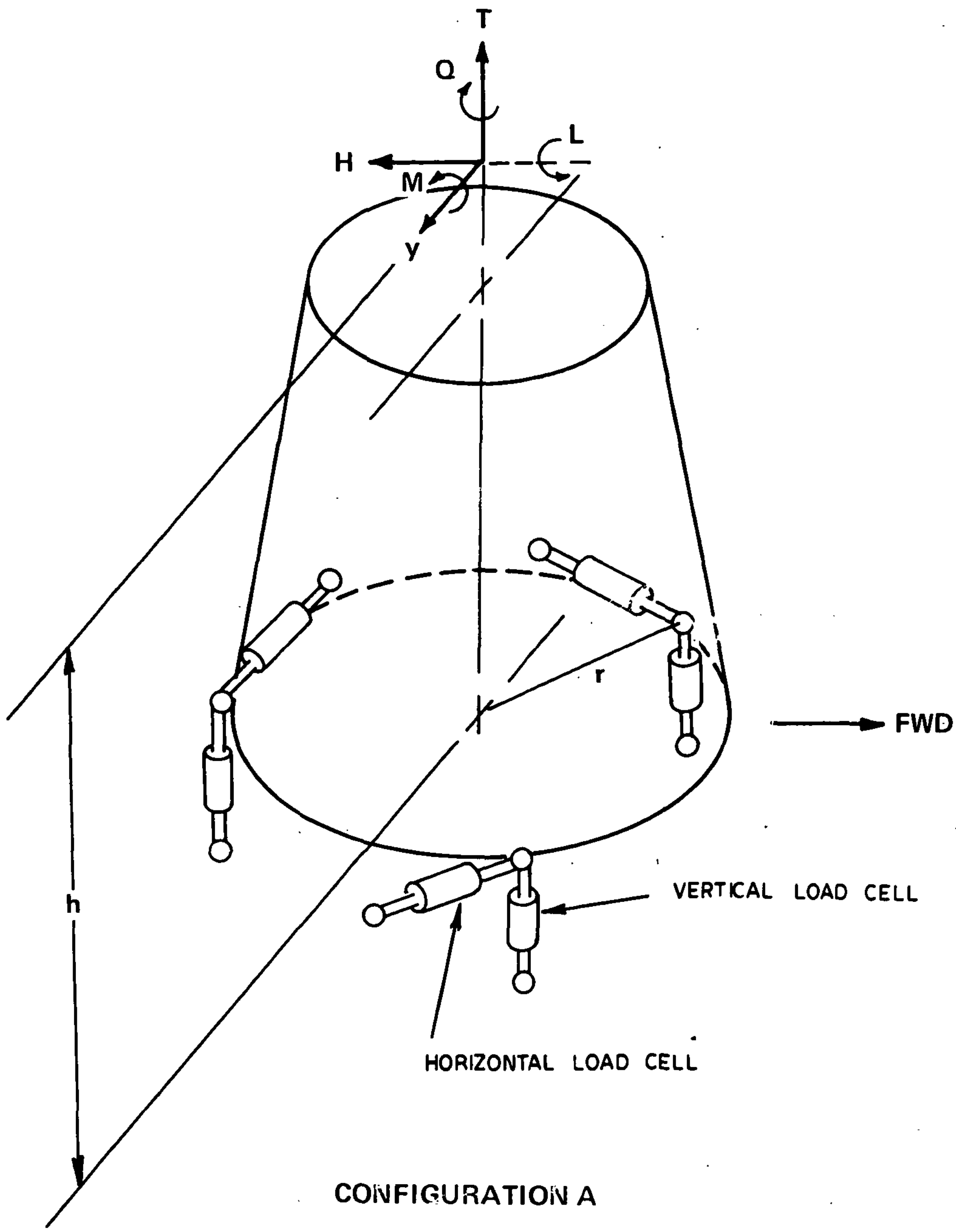
h = 5 5/6 ft.
r = 1 2/3 ft.
W_R = 1/5 G.W. (weight transmission system)

σ_i - appropriate standard deviation

Rotor forces as a function of transducer

Writing equilibrium equations for Configuration A yields the following:

$$
\begin{aligned}
H &= .866\, F_{2h} - .866\, F_{3h} \\
Y &= -F_{1h} + .5F_{2h} + .5F_{3h} \\
T &= -F_{1v} - F_{2v} - F_{3v} + W_R \\
L &= hF_{1h} - .5hF_{2h} - .5hF_{3h} + .866\, rF_{2v} - .866rF_{3v} \\
M &= rF_{1v} - .866yF_{2h} + .866\, YF_{3h} - .5rF_{2v} - .5rF_{3v} \\
Q &= rF_{1h} + rF_{2h} + rF_{3h}
\end{aligned}
\tag{1}
$$



CONFIGURATION A

FIGURE 92

ROTOR FORCE MEASUREMENT SYSTEM

Uncertainty in Rotor Forces

By basic statistical analysis:

$$\begin{aligned}
\sigma_H^2 &= (.866)^2 (\sigma_{2h}^2 + \sigma_{3h}^2) \\
\sigma_y^2 &= \sigma_{1h}^2 + (.5)^2 (\sigma_{2h}^2 + \sigma_{3h}^2) \\
\sigma_T^2 &= \sigma_{1v}^2 + \sigma_{2v}^2 + \sigma_{3v}^2 \\
\sigma_L^2 &= h^2 (\sigma_{1h}^2 + .5^2 \sigma_{2h}^2 + .5^2 \sigma_{3h}^2) + (.886r)^2 (\sigma_{2v}^2 + \sigma_{3v}^2) \\
\sigma_m^2 &= r^2 (\sigma_{1v}^2 + .5^2 \sigma_{2v}^2 + .5^2 \sigma_{3v}^2) + (.886r)^2 (\sigma_{2h}^2 + \sigma_{3h}^2) \\
\sigma_q^2 &= r^2 (\sigma_{1h}^2 + \sigma_{2h}^2 + \sigma_{3h}^2)
\end{aligned} \tag{2}$$

Uncertainty in Transducer

Assume

$$(1\sigma \text{ Transducer} = 1\% \text{ applied load} \pm 30 \text{ lbs}) \tag{3}$$

where applied load equals actual load in each transducer.

Transducer Forces

Rearranging equations (1) results in the transducer forces;

$$\begin{aligned}
F_{1h} &= (Q/r - 2Y)/3 \\
F_{2h} &- (Y + Q/r)/3 + .5H/.866 \\
F_{3h} &= (Y + Q/r)/3 + .5H/.866 \\
F_{1v} &= (W_R - T)/3 + 2(hH/r + M/r)/3 \\
F_{2v} &= (W_R - T)/3 + (hH/r + M/r)/3 + \tfrac{1}{2}(hY + L)/.866r \\
F_{3r} &= (W_R - T)/3 - (hH/r + M/r)/3 - \tfrac{1}{2}(hY + L)/.866r
\end{aligned} \tag{4}$$

Substitution of equation (4) into equation (3) then equation (3) into equation (2) yields the uncertainty in the rotor for any given test condition H, Y, T, L, M, Q.

APPENDIX B
CONTRACTOR PERFORMANCE VS NASA CR-114

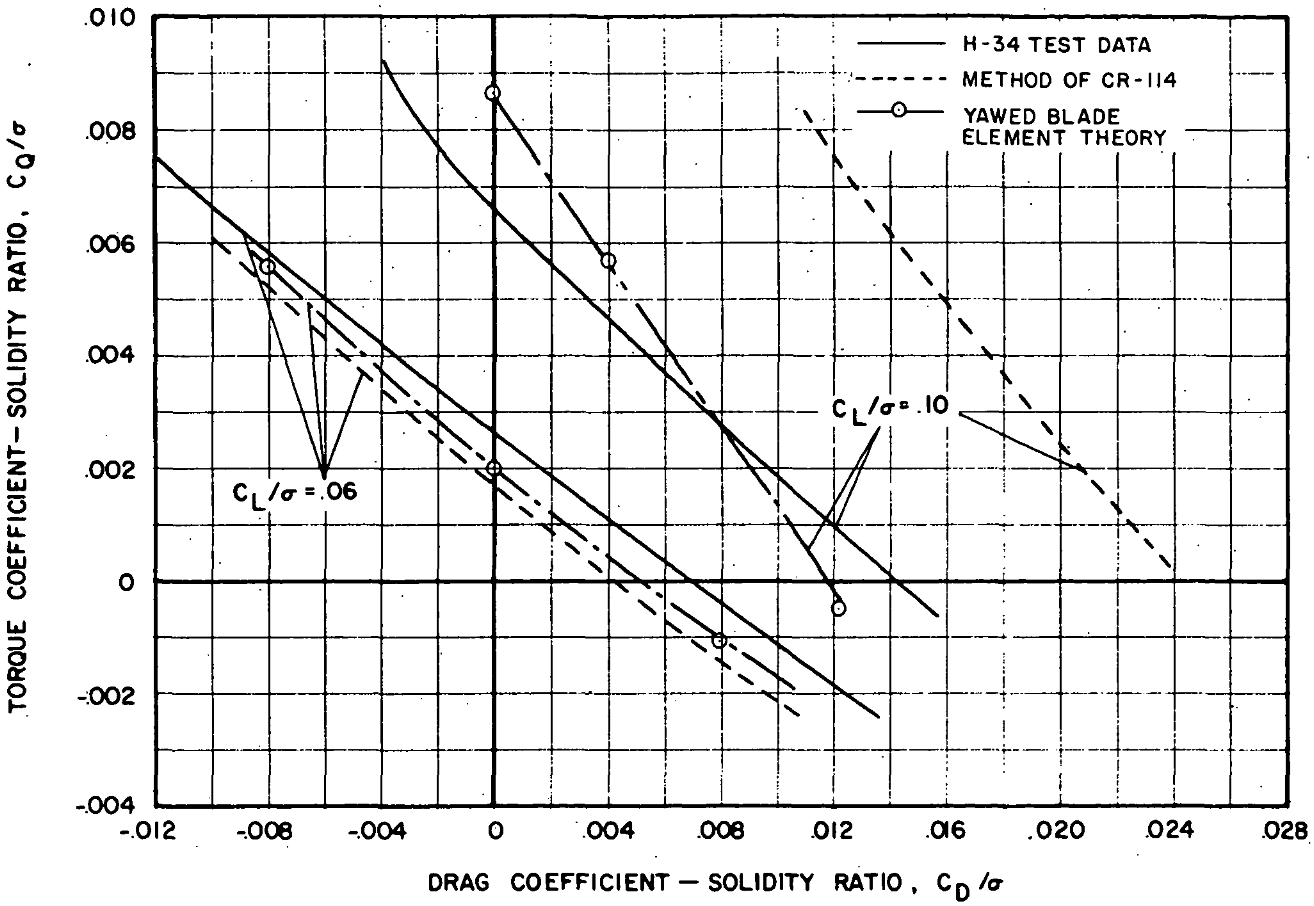
During Part I of the RSRA Predesign Study, a separate investigation was undertaken to compare the contractor's forward flight performance method with that of NASA CR-114. At the conclusion of Part I, the contractor recommended that its more conservative method be used. This approach was approved by the government. The following is a description of Sikorsky's Yawed Blade Element Rotor Performance method and a comparison of its high speed performance predictions with NASA CR-114.

Description of Yawed Blade Element Rotor Performance

This method is a forward flight performance analysis which is a logical out-growth of blade element theory (NASA CR-114). In this method, the component of local velocity acting along the blade span is not ignored as in the case of simple blade element theory. Instead, each blade element sees the entire local velocity vector acting at some yaw angle and at its true angle-of-attack, measured in the plane of the total velocity. Blade elemental lift and drag forces are then identified from 2-D airfoil data and applied in the local yawed coordinate system. Finally, the elemental yawed forces are integrated into total rotor lift, drag, and torque. Because there is no currently available yawed blade element airfoil data, normal airfoil data is used in computing the blade element lift and drag forces. Even with this approximation, better correlation with test data is achieved than with methods which entirely neglect the spanwise velocity component.

Stall Prediction

When using simple blade element theory, a general result noted at virtually all flight speeds is conservative stall prediction. That is, at higher rotor lift requirements, predicted power requirements far exceed measured values. An example of this behavior is given in Figure 93 showing the variation of rotor power with rotor drag, at constant lift. Note that at the theoretically stalled lift coefficient-solidity ratio of 0.1, the Yawed Blade Element theory (circled points) gives significantly better correlation with experiment than does normal theory (dotted lines). This improvement is believed to be a direct result of both the more realistic section velocity calculation and the more reasiltic calculation of section angle-of-attack for drag divergence afforded by the Yawed Blade Element method.



COMPARISON OF MEASURED AND PREDICTED TORQUE VARIATION WITH DRAG FOR BOTH A STALLED AND AN UNSTALLED VALUE OF LIFT.

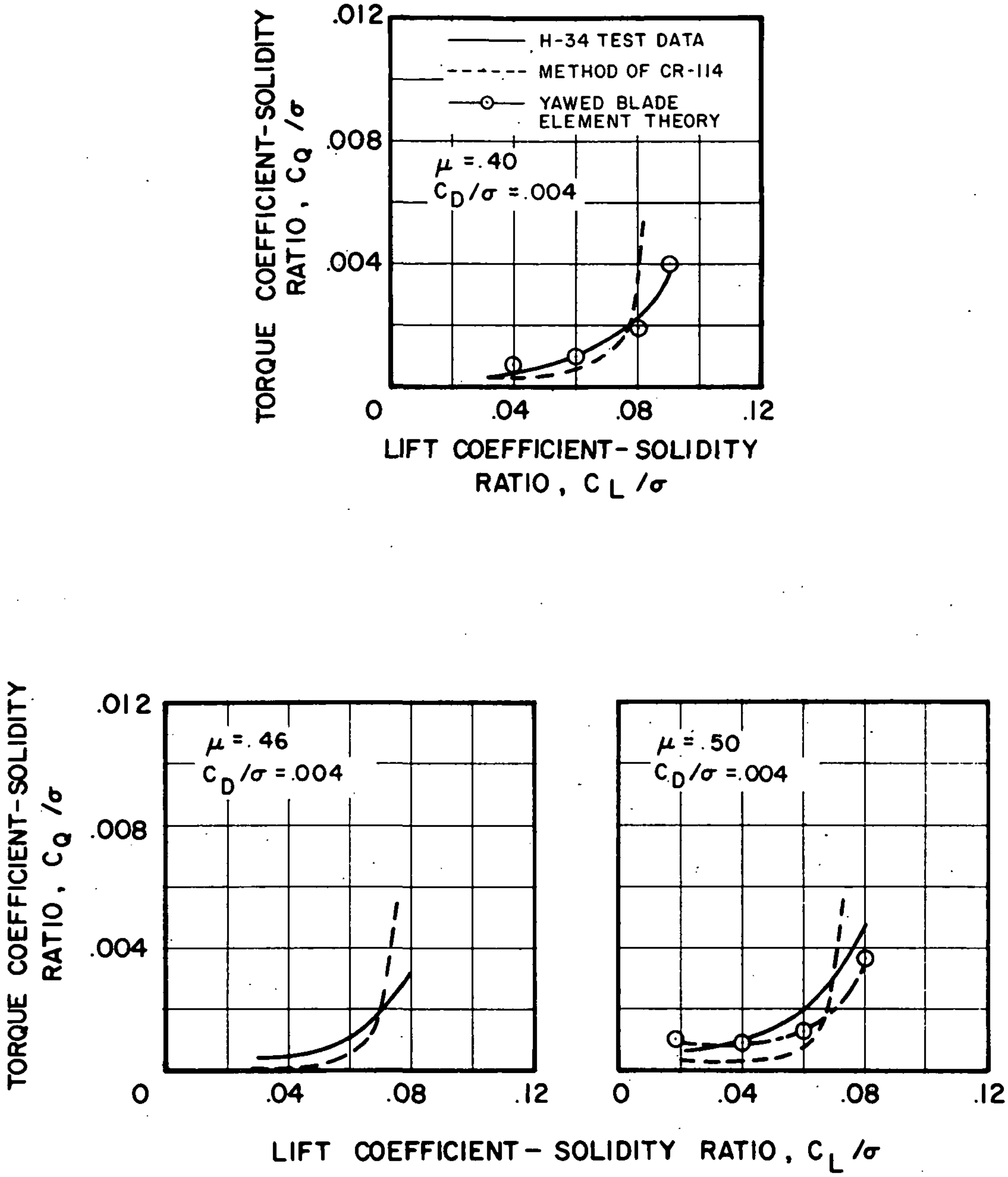
$\mu = .40$, $M_{(1.0,90)} = .67$

FIGURE 93

High Advance Ratio Performance Prediction

The tendency of standard performance calculations to become increasingly optimistic with increasing advance ratio is depicted in Figure 94 . Tip Mach number is approximately constant at 0.83 and a drag coefficient-solidity ratio of 0.004 is taken as being representative of other drag coefficients as well. Normal theory (dotted lines) is seen to underestimate measured power required (solid line) at low and moderate lifts, with this error becoming more apparent as advance ratio goes from 0.40 to 0.50. In all cases, high lift calculations with normal theory show premature power divergence.

Yawed Blade Element calculations are represented by the circled points at $\mu = 0.40$ and $\mu = 0.50$. These calculations show power coefficient to increase with advance ratio more in line with experimental trends. Also, power divergen at high lift continues to be better defined by the newer method, even at the higher advance ratio.



COMPARISON OF MEASURED AND PREDICTED TORQUE VARIATION WITH LIFT AT VARIOUS ADVANCE RATIOS.

$M_{(1.0,90)} \approx .83$

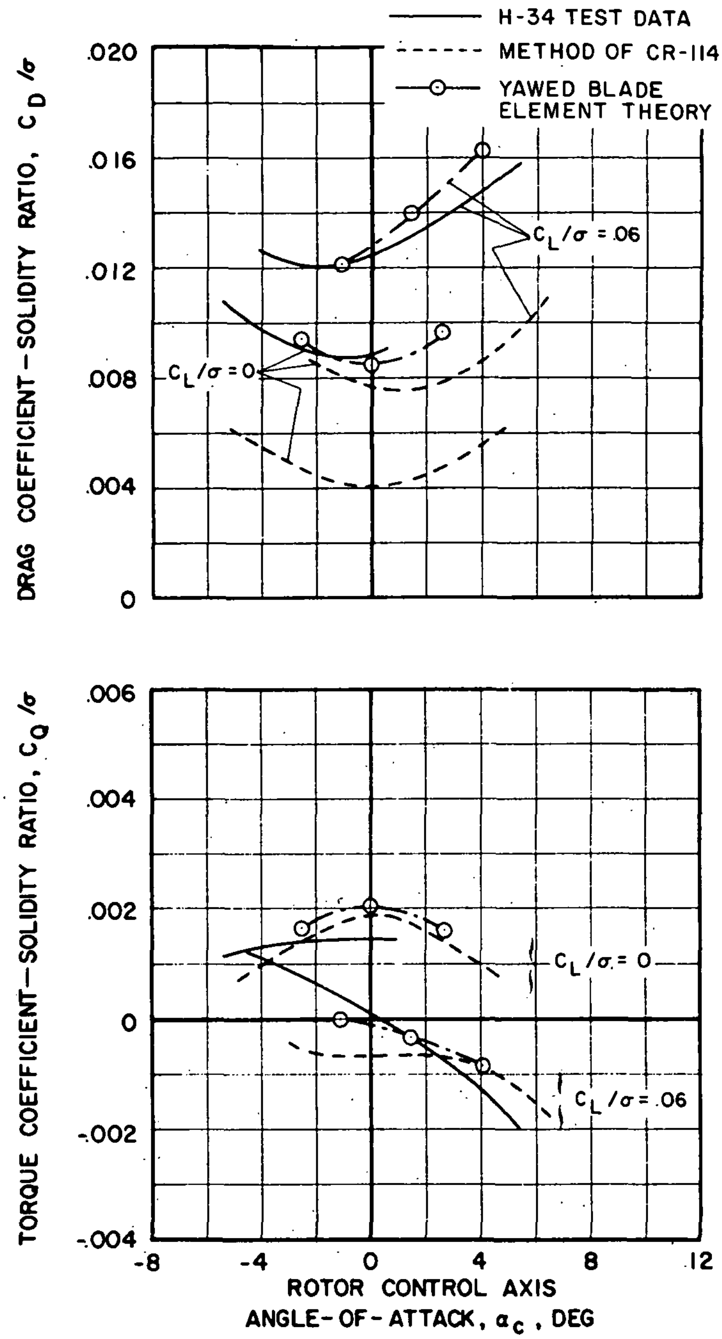
FIGURE 94

Figure 95 shows measured and predicted drag and torque coefficient at $\mu = 1.05$, plotted against rotor angle-of-attack (α_C). Two representative lift coefficients are shown, 0.06 and zero. It is seen that rotor drag coefficient is predicted by normal theory (dotted line) to be about 50 percent too low at both lift coefficients. The Yawed Blade Element calculations of drag coefficient (circled points) display a marked improvement in correlation with experimental data. This improvement in rotor drag prediction obtained with the Yawed Blade Element method is attributable to the inclusion in the theory of the spanwise component of blade element drag. As might be expected, the effect of this drag component on rotor performance becomes quite significant at high advance ratio due to the increase of local blade yaw angles.

With regard to torque calculations, it is seen that standard theory results bracket experimental results quite well as this low advancing tip Mach number (0.54). Yawed Blade Element calculated results are similar to standard results, with perhaps some qualitative improvement with the Yawed Blade Element method at $C_L/\sigma = .06$.

Sample RSRA Calculations

To give an indication of what might be predicted by the two methods for a typical RSRA condition, calculations were performed at 275 knots with a 6-bladed, 25-foot radius rotor having a solidity close to .1. The rotor was assumed lightly loaded (20%) with the shaft tilted back 2° to simulate compound helicopter operation. Calculation of the rotor equivalent L/D by the method of CR-114 yields a result approximately 20 percent higher than that predicted by Yawed Blade Element theory. Thus, in order to insure a conservative (and more accurate) theoretical approach at high advance ratio, and a better definition of stall at all advance ratios, the latter method should be used at selected design points to supplement the method of NASA CR-114.



COMPARISON OF MEASURED AND PREDICTED DRAG AND TORQUE VARIATION WITH ROTOR ANGLE-OF-ATTACK FOR A HIGH ADVANCE RATIO.

$\mu = 1.05$, $M_{(1.0,90)} = .54$

FIGURE 95

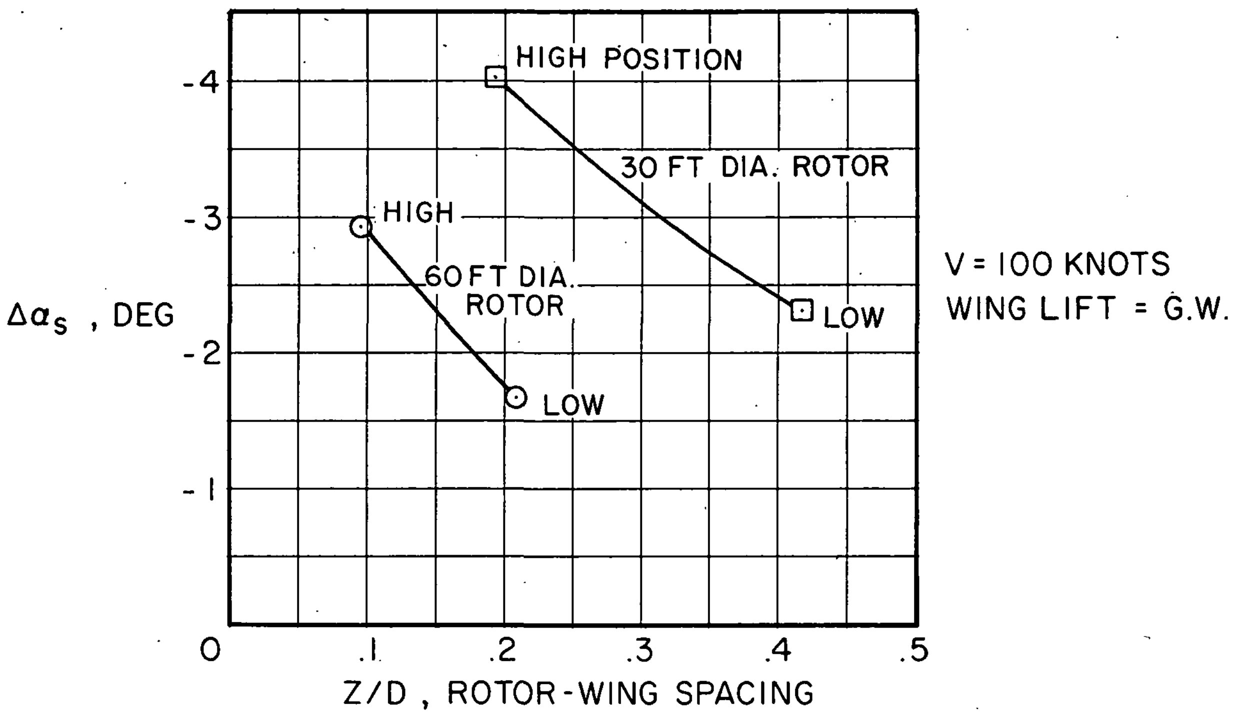
APPENDIX C
MUTUAL ROTOR/WING INTERFERENCE

The effect of the mutual interference of the wing and rotor was studied during Part I of the Predesign Study. In general, it was found that wing interference on the rotor was significant when the wing was used to produce lift to substantially unload the rotor. This interference decreased rapidly at higher (normal) rotor lift levels and as forward speed was increased at any lift level. It was also found that the influence of the rotor on the wing can be compensated for by varying wing incidence and aileron deflection. Specific areas of the interference study are described below.

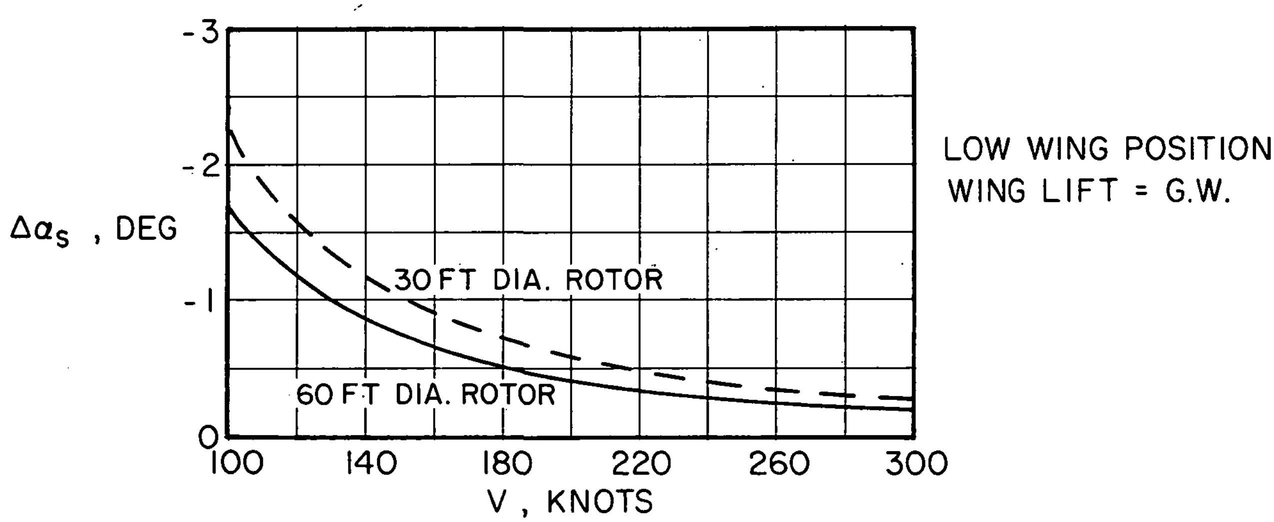
Effect of Wing Location

The area of prime concern in the interference study was the effect of the wing on the rotor at a given wing lift. A parameter was calculated which was defined as an average flow angle change at the rotor due to the wing. This parameter was called the "equivalent induced flow angularity." This equivalent induced flow angularity at the rotor was deduced from incremental rotor lift, rotor drag and from theory. Figure 96 shows the importance of wing vertical location expressed in terms of induced flow angularity for the condition with wing lift = gross weight and V = 100 kts. There is roughly a 2 to 1 ratio in angularity between the possible high and low wing locations. On this basis, the low wing position has been selected as most appropriate for the objectives of this program. Figure 96 illustrates the reduction in equivalent induced flow angularity with increasing speed at constant wing lift = gross weight. Since the angularity is directly proportional to wing lift coefficient, the illustrated condition wing lift = gross weight at 100 knots, is the most critical. As rotor lift is developed, the wing induced effects will diminish.

INTERFERENCE EFFECT OF THE WING ON THE ROTOR



(a) EFFECT OF WING LOCATION



(b) EFFECT OF FORWARD SPEED

FIGURE 96

Influence of the Wing on Rotor Lift and Drag

The influence of the wing on rotor lift is proportional to wing lift; thus it is a maximum when rotor lift tends to zero. The magnitude of this effect may be sufficient to cause lack of definition of the C_T/σ or θ relationship at low thrust levels. It may be possible to apply corrections to the test data if calibration tests are performed for the complete vehicle in the Ames wind tunnel. At high thrust levels, the interference effects will diminish. Figure 97 shows the influence of the wing on rotor lift and drag.

A rotor drag increment is due to rotation of the rotor lift vector by the wing downwash field. Incremental drag is then proportional to the product of wing and rotor lifts and a maximum when they are of equal magnitude. Data are shown for a rotor thrust/weight ratio of 0.6; the drag increment can be seen to diminish significantly with increased forward speed (i.e. with reducing C_L wing).

INFLUENCE OF THE WING ON ROTOR LIFT AND DRAG
LOW WING POSITION

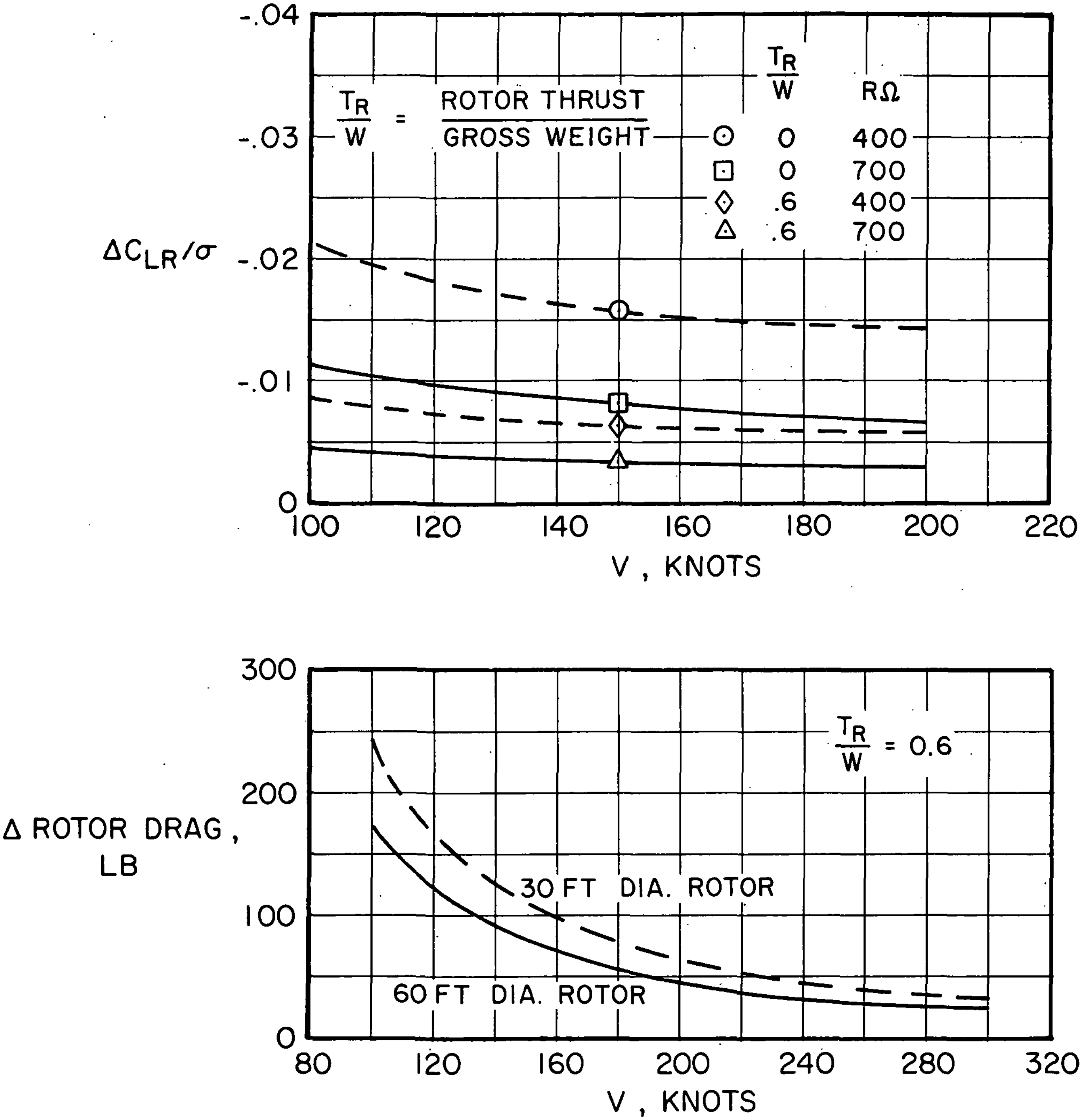


FIGURE 97

Influence of the Wing on Rotor Flapping

Figure 98 shows the resultant change in flapping induced by wing interference. The most significant effect is the relatively high lateral flapping induced at maximum wing lift (zero thrust on the rotor). The flapping effects are directly proportional to wing lift.

The induced lift and flapping effects have been generalized from the data presented in Reference 9 . The data were obtained theoretically by superimposing the calculated wing induced velocity field (including both the bound and trailing vorticity) upon the normal rotor inflow. The resultant forces, moments and flapping were then obtained using the same blade element theory used to generate the basic NASA CR-114. Since the model test rotor of Reference incorporated a δ_3 flapping hinge, the analysis was performed both with and without this feature. The test results correlated satisfactorily with prediction thus substantiating the method of analysis. The effect of δ_3, is to reduce the resultant flapping. For this study, in the interests of generality, no δ_3 coupling was assumed. It may be noted that the induced flapping values a_o and b_{1s} are directly proportional to Lock Number, a value of $\gamma = 12$ has been used for the presentation which is fairly typical of current articulated blades.

INFLUENCE OF THE WING ON ROTOR FLAPPING
LOW WING POSITION, $\gamma = 12$, 60 FT DIA. ROTOR

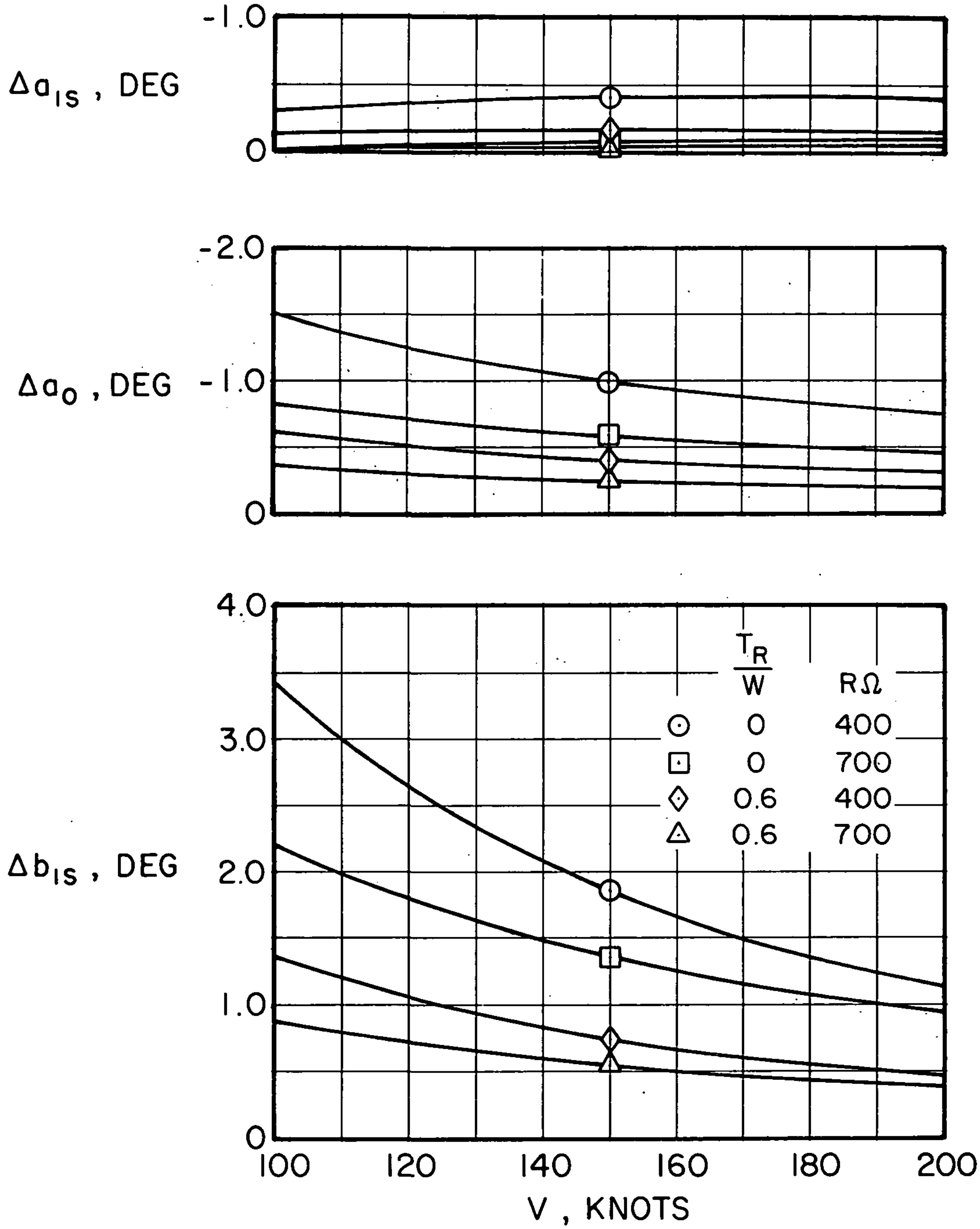


FIGURE 98

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